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NASA

PROJECT GEMINI

SUPPLEMENT

familiarization manual

SEDR 300 **COPY NO.**_

LONG RANGE and MODIFIED CONFIGURATIONS

THIS PUBLICATION SUPPLEMENTS
SEDR 300 VOLUME 1

MCDONNELL

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GUIDANCE and CONTROL SYSTEM

Section VIII

GROUP 4
Downgraded at 3 year intervals; declassified after 12 years

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GUIDANCE AND CONTROL - GENERAL

GENERAL

The Gemini spacecraft is equipped with highly advanced guidance and control systems. Five separate systems provide the guidance information and control capability required for precise attitude and velocity control. Guidance information can be either measured or computed as the occasion demands. The references utilized for guidance information are: inertial measurements, earth horizon, and time. Attitude control is provided about three (pitch, roll, and yaw) axes and is either manual or automatic as desired. A mode selector allows the pilot to select the type of control used. An attitude hand controller, located for use by either pilot, is utilized for manual attitude control. Velocity control is provided along three (longitudinal, vertical, and lateral) translational axes. A maneuver controller is utilized for manual velocity control. No provision is made for automatic velocity control. Information required by the pilot for manual attitude and velocity control is displayed by the appropriate guidance system. Guidance information and control capability for the non-rendezvous mission are provided by the following:

a. Attitude Control and Maneuver Electronics (ACME).

b. Inertial Guidance System (IGS).

c. Horizon Sensors

d. Time Reference Systems (TRS).

e. Propulsion System.

SYSTEM FUNCTIONS

The various guidance and control systems are all functionally related. The functional relationship between each of the systems is illustrated in Figure 8-1.

Attitude Control and Maneuver Electronics

The Attitude Control and Maneuver Electronics System converts input signals to thruster firing commands for the propulsion system. Input signals to ACME are provided by the attitude hand controller, the IGS, or the Horizon Sensors depending on the mode of operation.

Inertial Guidance System

The Inertial Guidance System provides inertial attitude and acceleration information, guidance computations, and displays. The inertial attitude and acceleration information is used for computations and display purposes. Computations are used for back-up ascent guidance, orbit correction and re-entry guidance. Displays are utilized by the crew for reference information and as a basis for manual control.

Horizon Sensors

Horizon Sensors provide a reference to the earth local vertical during orbit. Pitch and roll error signals are supplied to ACME for automatic attitude control and to the IGS for platform alignment.

Time Reference System

The Time Reference System provides a time base for all guidance and control functions. Time is displayed for pilot reference in both clock and digital form. The TRS also provides timing signals to the computer and the Sequential System.

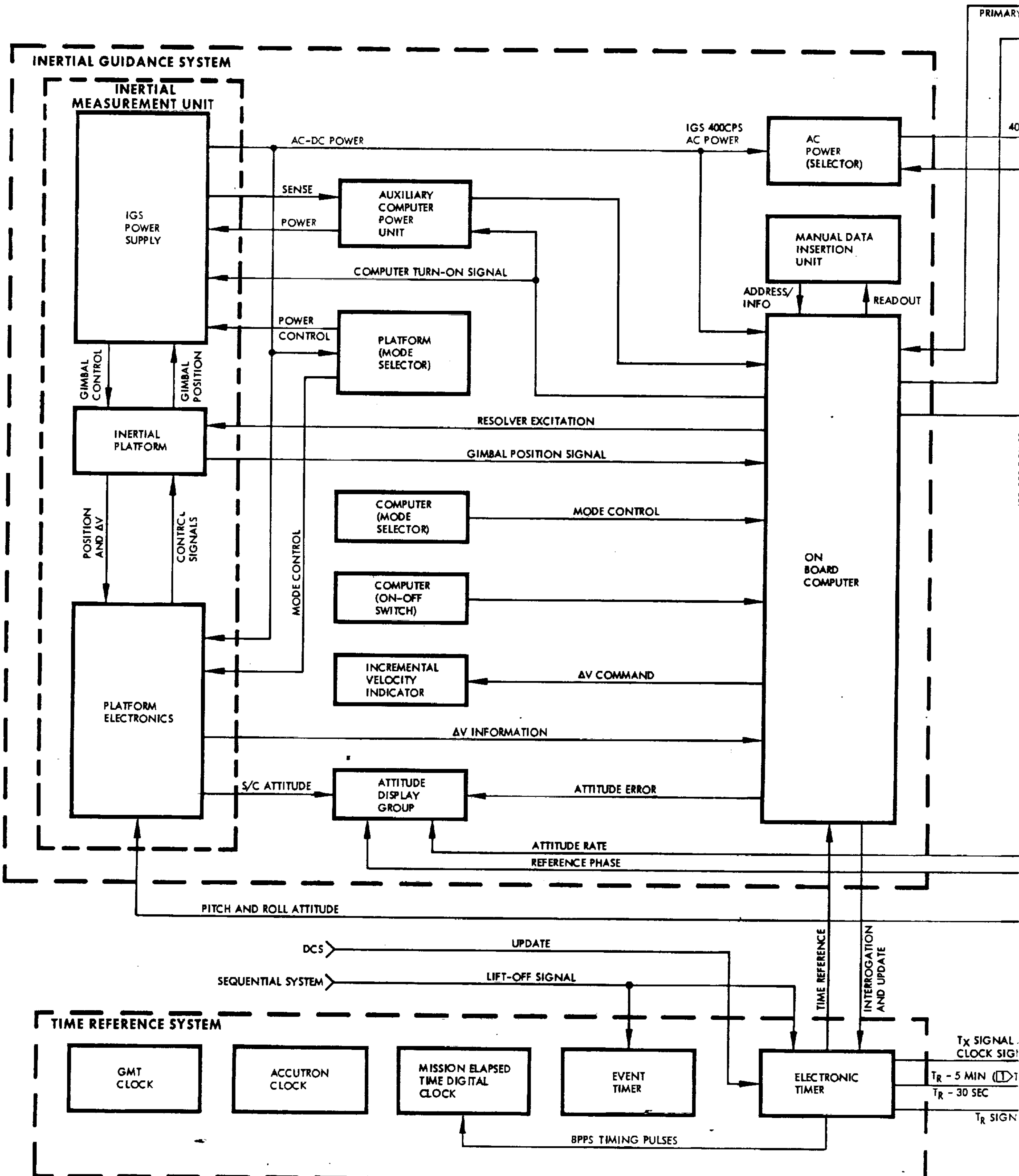


Figure 8-1 Guidance and Control Functional Block Diagram

Propulsion System

The Propulsion System provides the thrust required for spacecraft maneuvers. Thrusters are provided for both translational and attitude control. Firing commands for the Propulsion System are provided by ACME.

GUIDANCE AND CONTROL MISSION

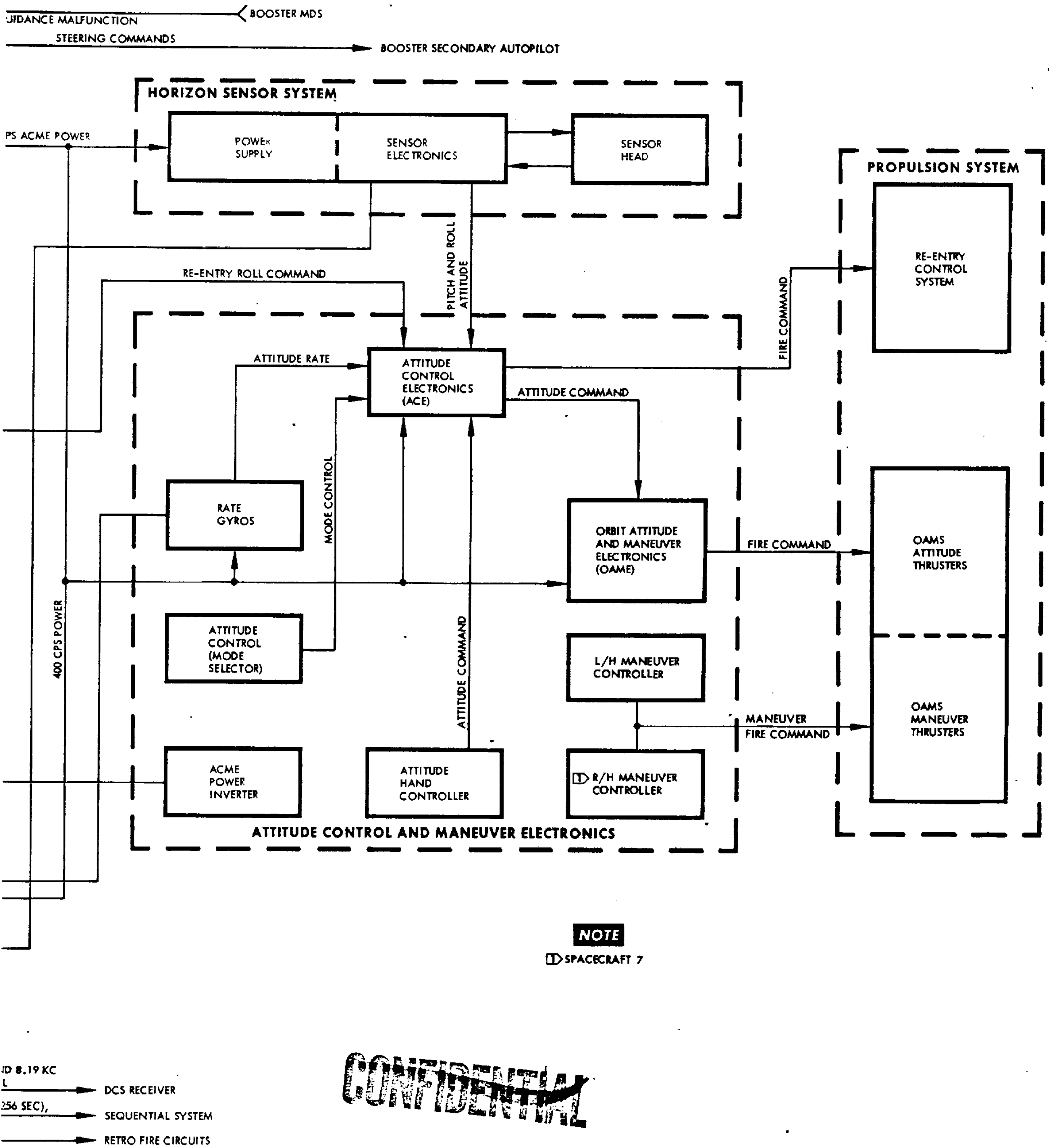
The functions of the guidance and control system are dependent on mission phase. The mission is divided into five phases for explanation purposes. The phases are: pre-launch, launch, orbit, retrograde, and re-entry.

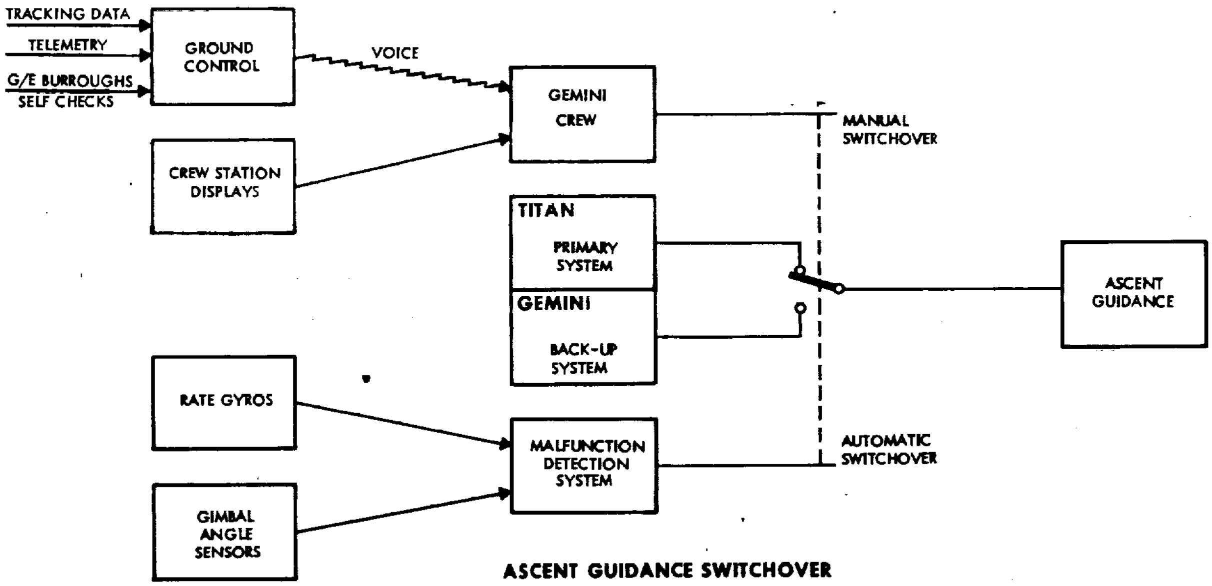
Pre-Launch Phase

Pre-launch phase is utilized for check-out and programming of guidance and control systems. Parameters required for insertion in the desired orbit are inserted in the computer. The IMU is aligned to the local vertical and the desired launch azimuth. Power is turned on to the various systems and mode selectors are placed in their launch position. Check-out and parameter insertion are performed in the last 150 minutes prior to launch.

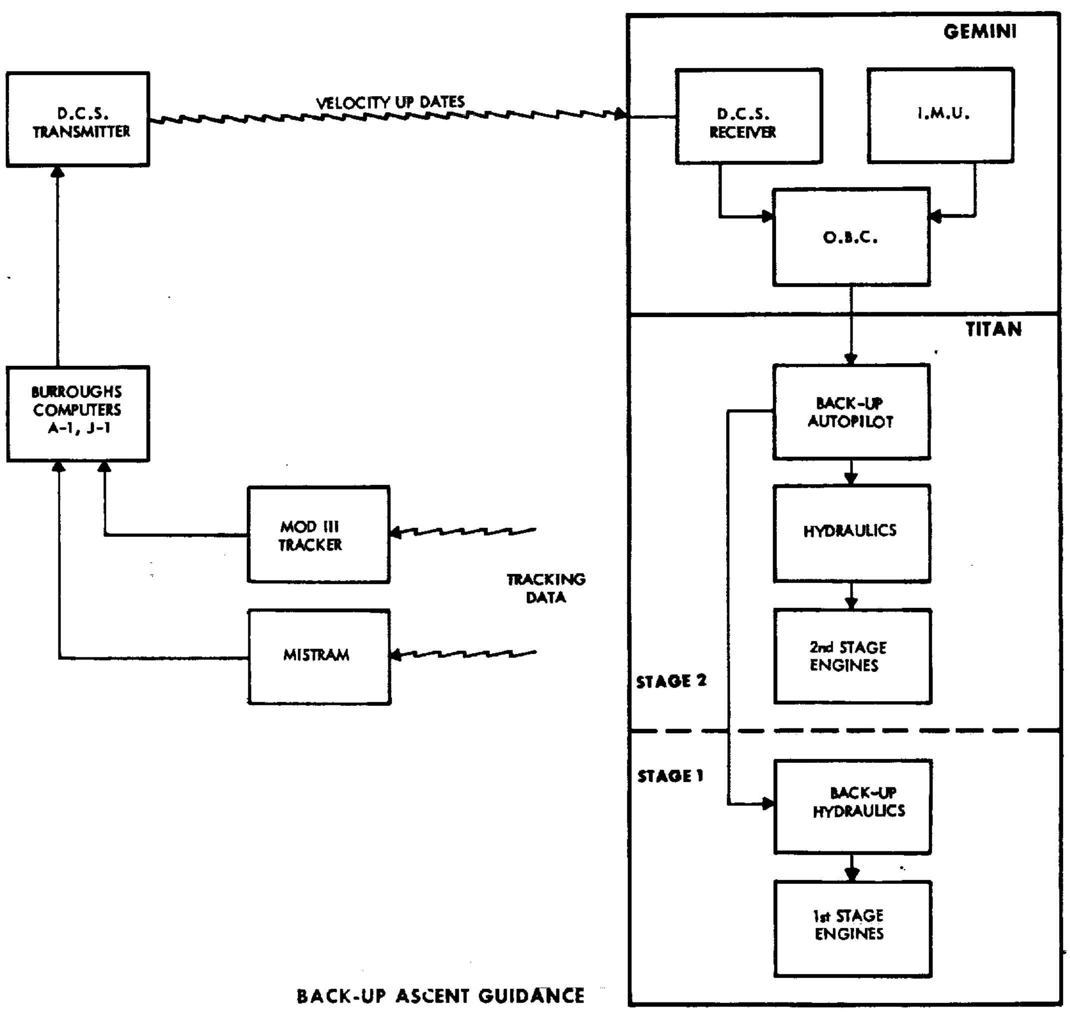
Launch Phase

Guidance and control from lift-off through SSECO is provided by the booster guidance system. However, in case of booster guidance malfunction the IGS can assume control. Provision is made for either automatic or manual switchover to back-up (Gemini) guidance. Figure 8-2 indicates both methods of switchover and the back-up method of controlling the booster during ascent. The IGS monitors attitude and acceleration parameters throughout the launch phase. Ground tracking information is used to continuously update computer parameters.


UIDANCE MALFUNCTION
BOOSTER MDS
STEERING COMMANDS
BOOSTER SECONDARY AUTOPILOT
HORIZON SENSOR SYSTEM
PS ACME POWER
POWER SUPPLY
SENSOR ELECTRONICS
SENSOR HEAD
PROPULSION SYSTEM
RE-ENTRY CONTROL SYSTEM
RE-ENTRY ROLL COMMAND
PITCH AND ROLL ATTITUDE
FIRE COMMAND
ATTITUDE RATE
ATTITUDE CONTROL ELECTRONICS (ACE)
ATTITUDE COMMAND
MODE CONTROL
RATE GYROS
ORBIT ATTITUDE AND MANEUVER ELECTRONICS (OAME)
FIRE COMMAND
OAMS ATTITUDE THRUSTERS
400 CPS POWER
ATTITUDE CONTROL (MODE SELECTOR)
ATTITUDE COMMAND
L/H MANEUVER CONTROLLER
MANEUVER FIRE COMMAND
OAMS MANEUVER THRUSTERS
ACME POWER INVERTER
ATTITUDE HAND CONTROLLER
R/H MANEUVER CONTROLLER
ATTITUDE CONTROL AND MANEUVER ELECTRONICS
NOTE
SPACECRAFT 7
ID 8.19 KC
L
DCS RECEIVER
256 SEC),
SEQUENTIAL SYSTEM
RETRO FIRE CIRCUITS



ASCENT GUIDANCE SWITCHOVER



BACK-UP ASCENT GUIDANCE

Figure 8-2 Gemini Ascent Guidance (Back-Up)

FM2-8-2

At SSECO, the remaining velocity required for insertion is displayed. The command pilot will, after separation, use the propulsion system to increase spacecraft velocity as required for insertion in the desired orbit. Insertion will take place approximately 580 miles down range at an inertial velocity of approximately 25,770 feet per second.

Orbit Phase

Orbit phase is utilized for checkout and alignment of systems, orbital maneuvers, experiments and preparation for retrograde and re-entry. Immediately after insertion a series of system checks will be performed to assure the capability of guidance and control systems. Guidance computations and measurements are checked for accuracy against ground tracking information. Systems are updated and aligned by ground command (DCS) or by the pilot. After completion of system checks, the orbital maneuvers and experiments can be performed. During the final orbit, guidance and control systems are re-aligned in preparation for retrograde and re-entry.

Retrograde Phase

Retrograde phase begins approximately five minutes before retrofire. The computer is placed in re-entry mode and begins collecting data for re-entry computations. The Time Reference System provides indications at T_R-5 minutes (T_R-256 seconds on spacecraft 7), T_R-30 seconds, and T_R. At T_R-5 minutes or T_R-256 seconds (depending on spacecraft number) a minus 16 degree bias is placed on the pitch attitude needle. The Propulsion System is switched from orbit attitude and maneuver to re-entry control. Spacecraft attitude is controlled manually during retrograde. Retrograde acceleration and attitude are monitored by the IGS and velocity changes are displayed for reference.

Re-Entry Phase

Re-entry phase begins immediately after retrofire. The event timer counts through zero at retrograde and will be counting down from one hundred minutes (60 minutes on spacecraft 7) during re-entry phase. After retrofire the retrograde adapter and horizon scanner heads are jettisoned. Shortly after retrograde, the pilot orients the spacecraft to re-entry attitude (0° pitch, 180° roll, 0° yaw). Re-entry attitude is held until the computer re-entry program starts. At approximately 400,000 feet altitude, the computer re-entry program starts and the pilot has a choice of manual or automatic control. For manual control, the pilot selects RE-ENT RATE CMD or for automatic control, the RE-ENT mode is utilized. In the automatic mode, the computer controls spacecraft roll attitude. For either mode of control, the flight director is referenced to the computer and indicates computed attitude commands. The purpose of the computer re-entry program is to control the point of touchdown and control re-entry heating. By controlling the spacecraft roll attitude and rate, it is possible to change the down range touchdown point by approximately 300 miles and the cross range touchdown by 25 miles left or right. The relationship between roll attitude or rate and direction of lift is illustrated in Figure 8-3. The roll control starts at approximately 400,000 feet and ends at 90,000 feet. Re-entry phase ends at 80,000 feet when the computer commands an attitude suitable for drogue chute deployment.

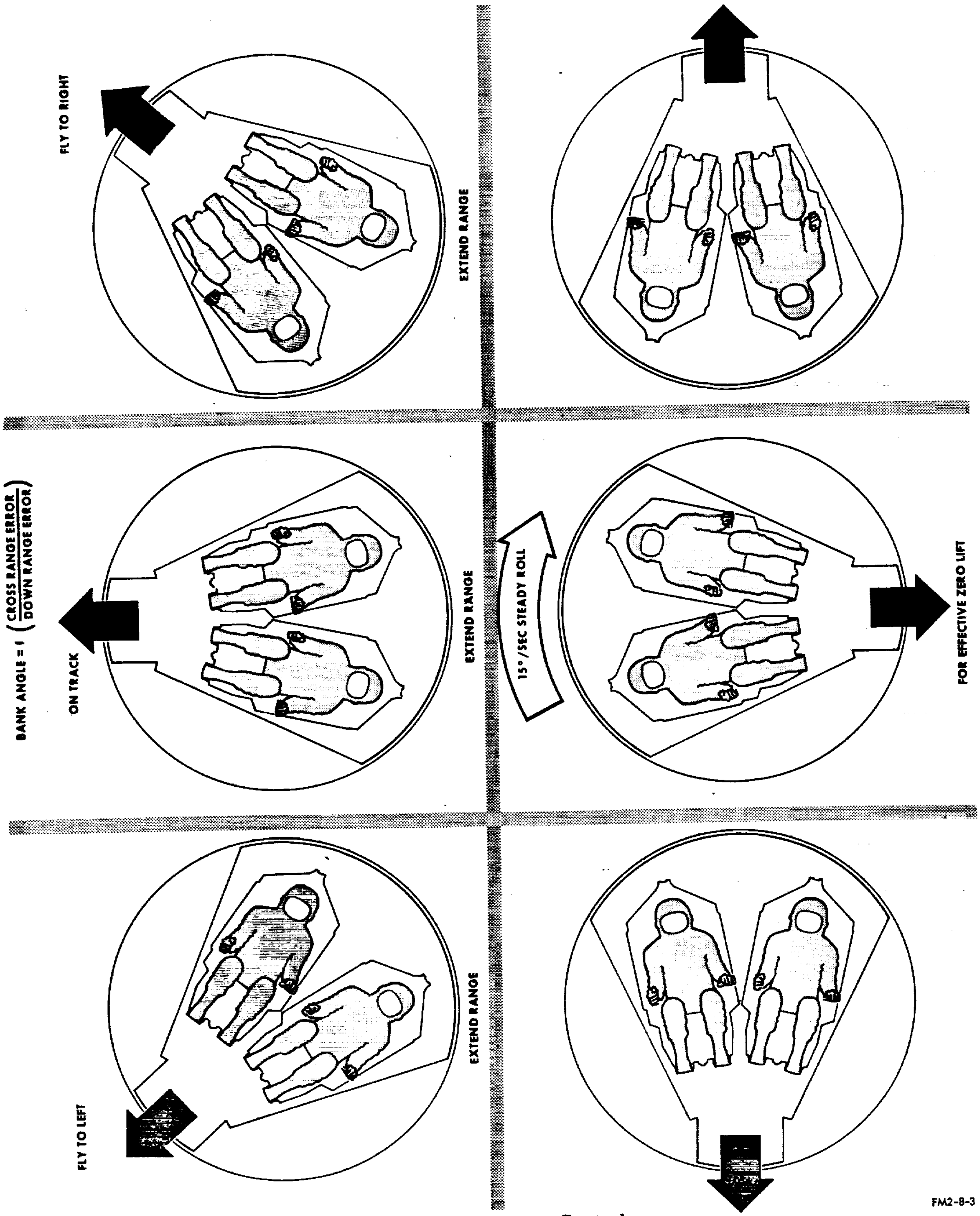


Figure 8-3 Re-entry Control

ATTITUDE CONTROL AND MANEUVERING ELECTRONICS

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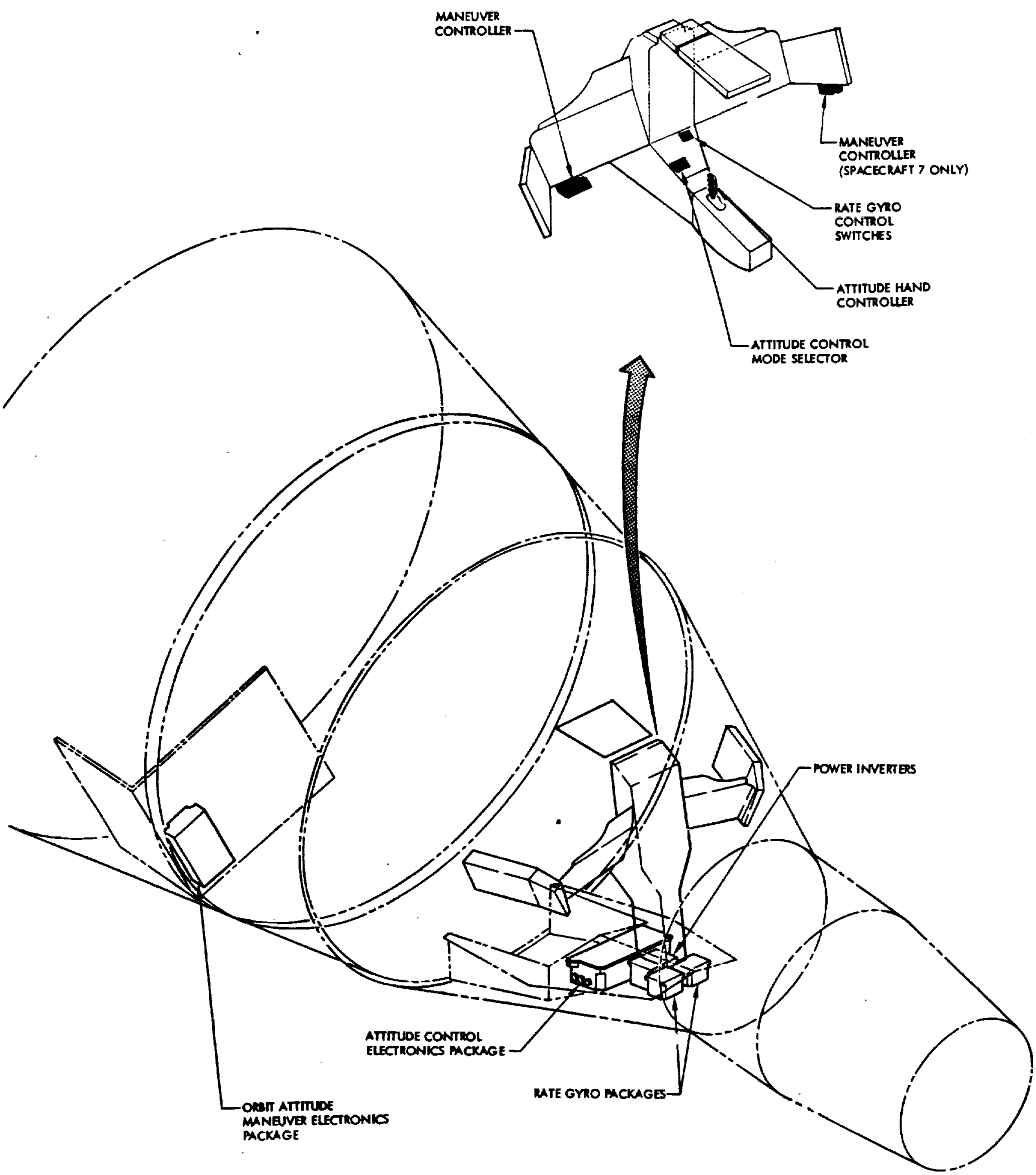


Figure 8-4 Attitude Control and Maneuver Electronics

ACME SYSTEM

SYSTEM DESCRIPTION

The Attitude Control and Maneuver Electronics (ACME) System (Figure 8-4) provides the control circuitry to attain and/or maintain a desired spacecraft attitude or velocity. The ACME accepts signal inputs from the attitude hand controller, maneuver hand controller, horizon sensors, platform or the computer; processes the signal, and applies a firing command to the appropriate Propulsion System solenoid valves. ACME is composed of four separate sub-systems: Attitude Control Electronics (ACE), Orbit Attitude and Maneuver Electronics (OAME), a Power Inverter and two identical Rate Gyro Packages. The ACE, power inverter and rate gyro packages are installed in the center bay of the re-entry module. The OAME package is located in the equipment section of the adapter. Total weight of the ACME System is approximately 40 pounds.

The ACME provides the capability of automatic or manual attitude control, with seven separate, selectable modes of operation. The horizon sensor, the inertial platform or the computer provide the reference for automatic modes of operation. The attitude hand controller provides the input signals for manual modes of attitude control, and the maneuver hand controller provides input signals for translational maneuvers.

SYSTEM OPERATION

GENERAL

The ACME provides attitude control, automatic or manual, during all flight phases of the spacecraft mission. Rate gyro inputs to ACE are used to damp spacecraft attitude rates. Signal inputs are modified by ACME logic and converted into fire commands for the propulsion system.

The ACME functional modes of control are horizon scan, rate command, direct, pulse, re-entry rate command, re-entry and platform. Each mode provides a different signal input (or combination of inputs) to be processed by ACE for routing to RCS or OAME solenoid valve drivers. The modes of control are separated into two basic types: automatic attitude control modes (horizon scan, re-entry and platform) and manual attitude control modes (rate command, direct, pulse and re-entry rate command). Display information from control panel indicators is used as reference when manual control modes are selected. Reference information for manual control is supplied by guidance and control sub-systems, and consists of the following: Attitude, attitude rates, bank angle and roll commands (from the attitude display group) and velocity increments (from the incremental velocity indicator). The control panels also contain the control switches necessary for selection of ACME power and logic circuits and mode of attitude control, along with selection switches for the various ACME redundant options.

FUNCTIONAL OPERATION (ACME)

Attitude Control (See Figure 8-5)

Commands or error signals from the computer, platform, horizon sensors, rate gyros and attitude hand controllers are converted by the ACE into thruster firing commands. The firing commands are routed by a valve driver select system to the RCS or the OAMS attitude solenoid valve drivers.

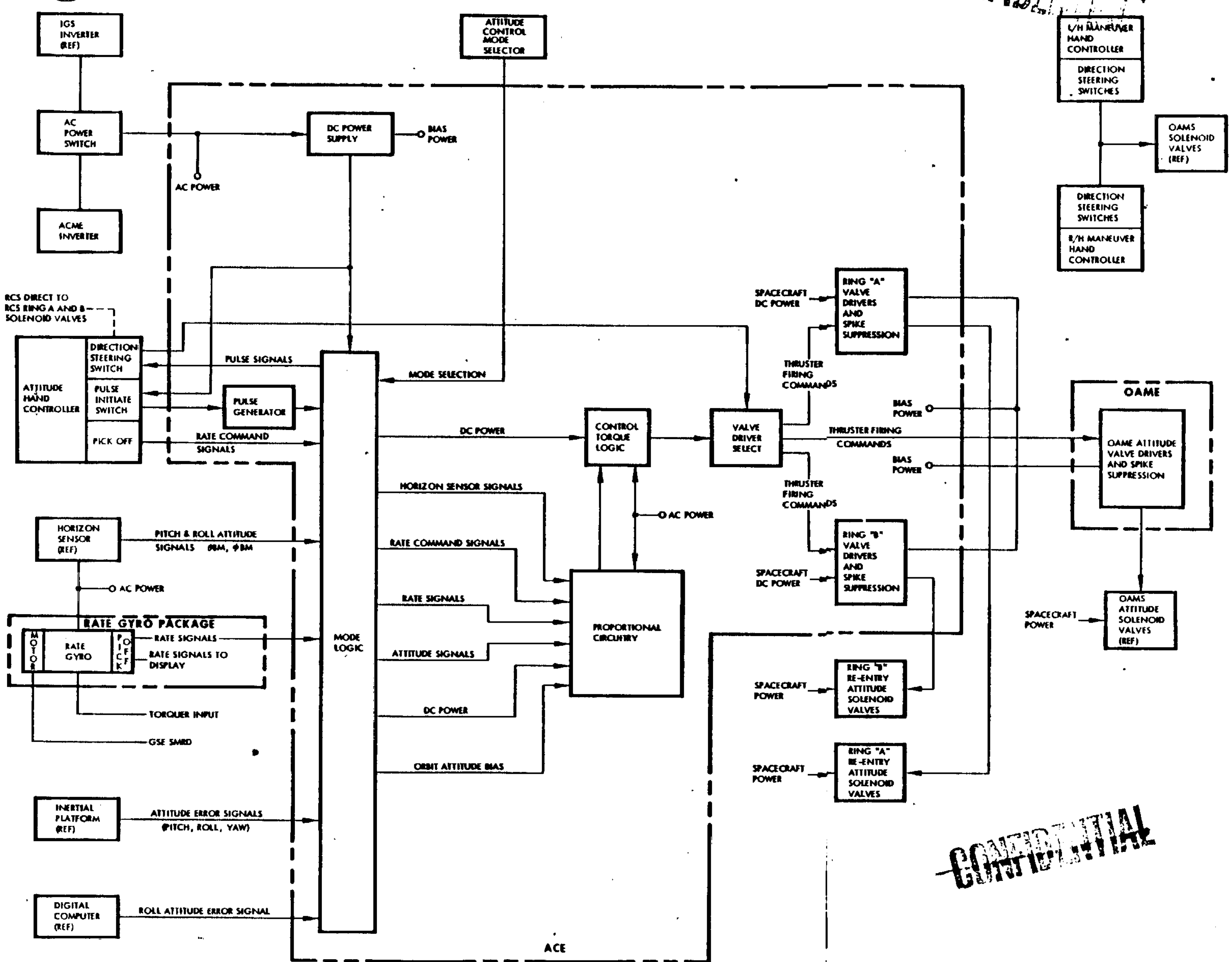


Figure 8-5 ACME Functional Block Diagram

FM2-8-5

Signal inputs to the ACE are of three types: AC attitude signals, DC attitude signals and AC attitude rate signals. These signals are selected and distributed by ACE mode logic switching circuits. Selected signals are channeled through the proportional circuitry which amplifies, sums and demodulates the signal inputs into a DC analog output. Horizon Sensor (DC attitude) signals are converted to AC prior to entering the proportional circuitry. The analog signals are then converted by control torque logic switch circuitry to a positive or negative discrete, the output consisting of either positive or negative thruster firing commands. These commands are routed by the valve driver select circuit to the RCS, (ring A and/or ring B) valve drivers, or to the OAMS attitude valve drivers for a fire command to the appropriate thruster valves. Zener diode spike suppression circuits, limit the voltages generated across the solenoid valves during current interruptions.

Attitude Hand Controller

Spacecraft attitude may be manually controlled by use of the attitude hand controller and a visual reference. Controller outputs are rate, pulse or direct command signals, (plus a hand controller position output to telemetry) depending on the control mode selection. Output signals are produced by handle movements, about each respective axis, from the centered position. Rate signals produced are proportional to the amount of control displacement from a center deadband. Direct and/or pulse signals are produced when the hand controller is displaced past a preset threshold or deadband. Pulse signals trigger a calibrated on time of a pulse generator in ACE. The control handle must be returned to a neutral position before another single pulse can be commanded. Details

of each mode of control may be found in the mode operation paragraph.

RCS Direct

The RCS direct mode is selectable as an alternate means of manually firing the RCS thrusters, and by-passes the ACE. The DIRECT position of each of the RCS RING A and/or RING B switches provides a circuit ground to 12 attitude hand controller RCS direct switches. The ground is then applied directly to the required thruster solenoid valves through appropriate hand controller displacements. This RCS mode of operation is intended for standby or emergency control only.

Maneuver Hand Controller

Translational maneuvers of the spacecraft, in the horizontal, longitudinal and vertical planes, are commanded by the maneuver hand controller. Displacement of the controller from the centered or neutral position to any of the six translational directions produces a direct on command to the respective solenoid valve drivers.

Rate Gyros

The function of the rate gyro package is to sense angular rate about the pitch, yaw and roll axes of the spacecraft and provide an output signal proportional to that sensed rate. Selection of certain control modes provides gyro inputs to ACE for angular rate damping. Additional information concerning the rate gyros may be found in the paragraph on system units.

Power Inverter

The power inverter provides the ACME and horizon sensors with AC power. Spacecraft DC power is converted to 26V, 400 cps. (The IGS inverter provides the primary source of AC excitation.) The ACME inverter is utilized when the inertial measuring unit is not operating. Additional information on the power inverter may be found in the paragraph on system units.

MODE OPERATION

Control of spacecraft attitude is accomplished through the selection of seven functional modes of control. Each mode of control is utilized for a specific purpose or type of ACME operation in conjunction with various mission phases. Each mode of operation provides either automatic or manual spacecraft control through the switching of input signals to ACE. In addition, the mode logic circuits de-energize all unused circuits within the ACE during use of the horizon scan mode to conserve power. Switching is performed by transistors at the signal level and by relays at the power level. The operation of each mode of control is explained in the following paragraphs.

Direct Mode (M_1)

In this mode, thruster firing commands are applied directly to the RCS or OAME attitude solenoid valve drivers, by actuation of the attitude hand controller direct switches (Figure 8-6). Selection of the DIRECT mode applies an ON bias voltage to a transistor designated ground switch A. Conduction of the transistor completes a circuit to ground which is common to one side of the hand controller direct switches. The transistor remains on as long as the direct

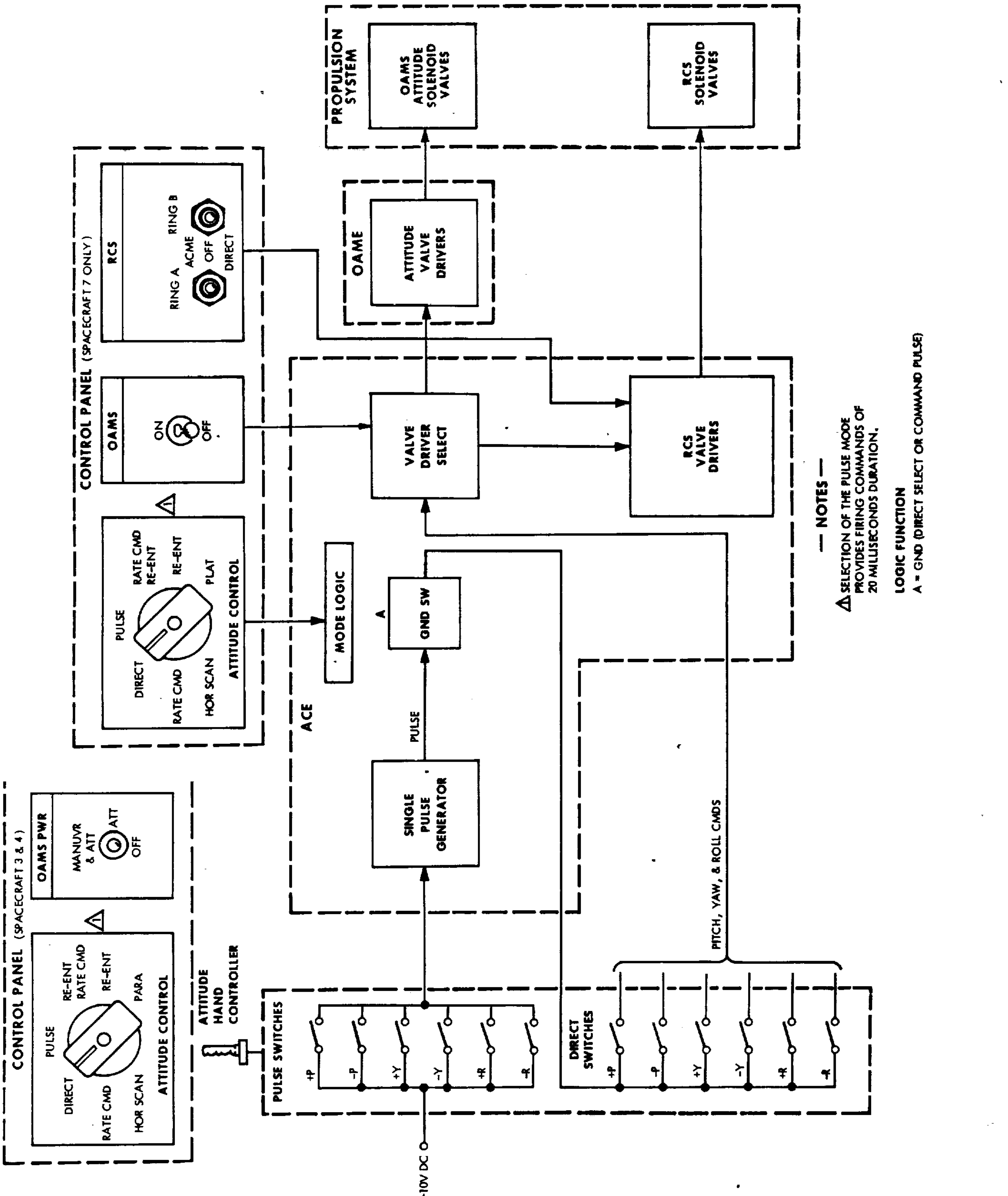


Figure 8-6 ACME Simplified Block Diagram (Direct & Pulse Command Modes)

FM2-8-6

mode is selected.

Three sets of six normally open switch contacts provide the command signals in the pitch, yaw and roll axis and will close when the hand controller is moved beyond a preset threshold (2.5 degrees) of handle travel. Movement in the desired direction applies a ground from switch A directly to the valve driver relative to that direction and in turn fires the proper thruster(s). Thrusters continue firing as long as the hand controller is displaced beyond the 2.5 degree threshold. This mode of operation is optional at all times.

Pulse (M2)

In this mode, the attitude commands initiated by hand controller displacement fire a single pulse generator in the ACE (Figure 8-6). The pulse mode energizes the generator, allowing it to fire for a fixed duration when a pulse command is received. Commands originate every time one of the six normally open pulse switch contacts of the hand controller is closed. This triggers the generator and applies a bias voltage pulse for a 20 millisecond ON duration to ground switch A. This ground is then applied to the RCS or OAME attitude valve drivers, through the actuated hand controller direct switches as a command for thruster firing. Commands may be initiated in the pitch, yaw or roll axis by moving the control handle in the desired direction beyond a preset threshold (3.5 degrees). Thrusters fire for 20 milliseconds each time the handle is displaced beyond 3.5 degrees. This mode is optional at all times and will normally be used during platform alignment.

Rate Command Mode (M3)

In this mode, spacecraft attitude rate about each axis is proportional to the attitude hand controller displacement from the neutral deadband (Figure 8-7). (Pickoff excitation is zero for displacements less than 1 degree of handle travel, providing a non-operational area or deadband.) Command signals, generated by handle displacements, are compared to rate gyro outputs and when the difference exceeds the damping deadband, thruster firing occurs. Signals originate from potentiometers in the hand controller and outputs are directly proportional to handle displacement. A maximum command signal to ACE produces an angular rate of 10 degree/second about the pitch and yaw axis and 15 degrees/second about the roll axis.

Automatic, closed loop stabilization of spacecraft rates is provided from the sensing of angular rates by the rate gyro package. With the absence of hand controller command signals, spacecraft rates about each axis are damped to within ± 0.2 degrees/second with OAME attitude control and to within ± 0.5 degree/second with RCS attitude control. Output signals from the rate gyros are used to produce fire commands until the rate signal is within the damping deadband. This mode is optional at all times and will normally be used during translational thrusting or attitude changes.

Horizon Scan Mode (M4)

In this automatic command mode, horizon sensor outputs (pitch and roll) are processed by the ACE to orient and hold the spacecraft within a desired attitude deadband during orbit (Figure 8-8). Pitch attitude is maintained automatically to within ± 5 degrees of the -5 degree reference and roll attitude is maintained

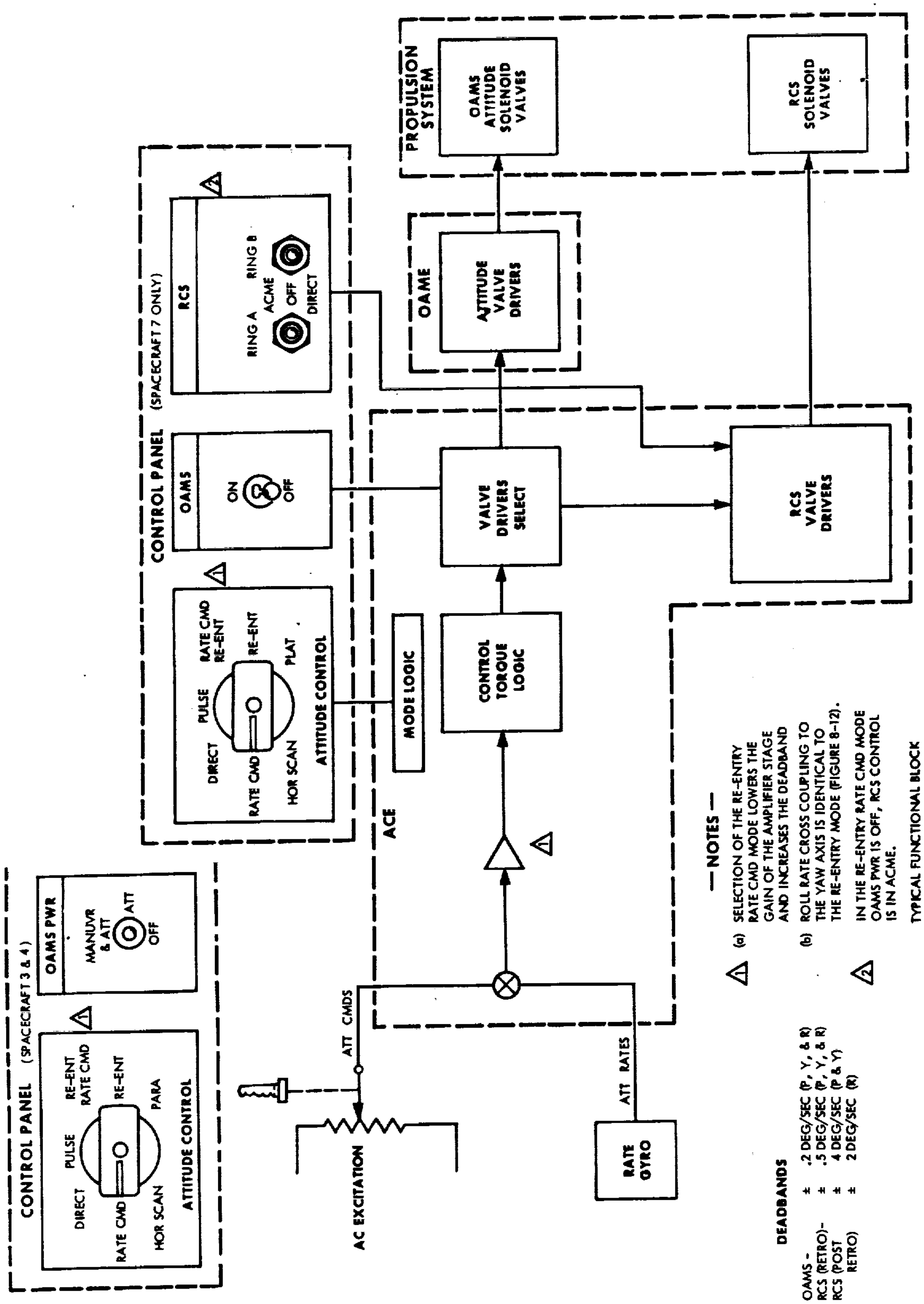


Figure 8-7 ACME Simplified Block Diagram (Rate Cmd. and Re-entry Rate Cmd. Modes) FM2-8-7

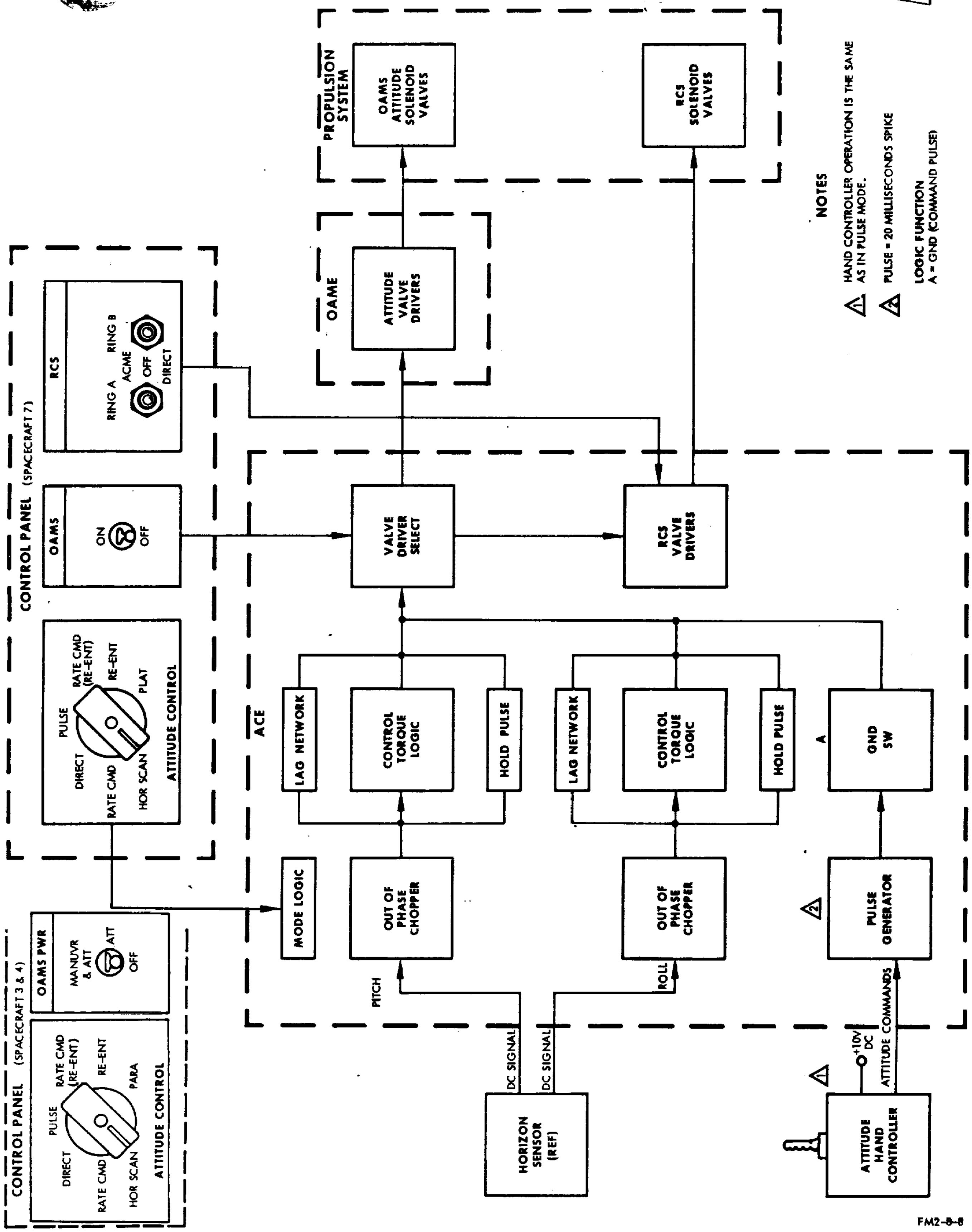


Figure 8-8 ACME Simplified Block Diagram (Horizon Scan Mode)

automatically to within ±5 degrees of the zero degree null. Control about the yaw axis is accomplished by command from the attitude hand controller, in the same manner as in the pulse mode. Pulse control about the pitch and roll axes is also available to supplement automatic control. A bias voltage is summed with the horizon sensor pitch output to maintain the 5 degree pitch down orientation. When the attitude error (pitch or roll) exceeds the 5 degree control deadband, the output of the ACE on-off logic is a pulse firing command. The pulse on time is for 18 milliseconds and the pulse repetition frequency is dependent upon how much the attitude error exceeds the 5 degree deadband. A lag network in this mode provides a pseudo rate feedback for rate damping, without having to use the power consuming rate gyros.

Re-entry Mode (M5)

In this automatic command mode, spacecraft angular rates about the pitch and yaw axes are damped to within ±5 degrees/second and to within ±2 degrees/second about the roll axis (Figure 8-9). Roll attitude is controlled to within ±2 degrees of the attitude commanded by the digital computer input to ACE. Computer roll input to ACE consists of either a bank angle attitude command or a fixed roll rate command, depending on the relationship between the predicted touchdown point and the desired touchdown point. When a roll rate is commanded, roll to yaw crosscoupling is provided to minimize the spacecraft lift vector.

Re-entry/Rate Command Mode (M5D)

In this manual command mode, spacecraft rates are controlled by rate commands from the attitude hand controller. The method is identical to the rate command

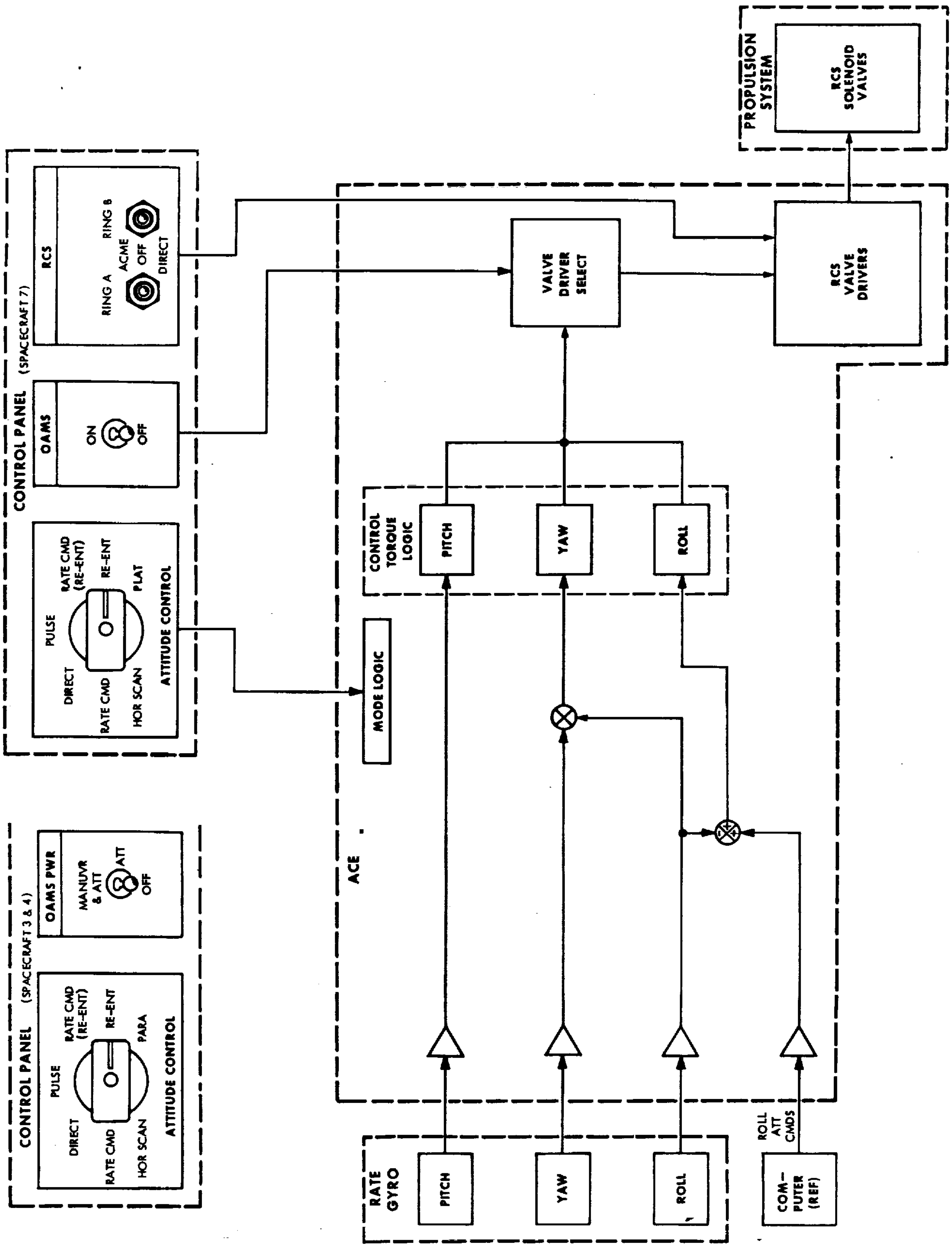


Figure 8-9 ACME Simplified Block Diagram (Re-entry Mode)

FM2-8-9

mode with the addition of roll-yaw rate crosscoupling. Angular rate damping about the three axes is identical to the re-entry mode. The computer bank angle and roll rate commands do not automatically control the spacecraft but are provided on the control panel displays where they can be used as a reference for initiating manual re-entry roll commands.

Platform (M6)

This attitude control mode is used on spacecraft 7 to maintain a fixed attitude in all three axes, with respect to the inertial platform. Spacecraft attitude is held automatically to within 1.1 degrees of the platform attitude. A horizontal attitude, with respect to the earth, can be held if the inertial platform is in the orbit rate or alignment modes of operation. Spacecraft attitude rates are damped to within 0.5 degrees/second. The primary purpose of this mode is to automatically hold an inertial spacecraft attitude. PLAT mode is also useful for maintaining spacecraft attitude during fine alignment of the platform. (See Figure 8-10.)

Aborts - ACME/RCS

Rate command mode of ACME will be utilized for attitude control during all abort modes. Control over the RCS Ring A and Ring B switches, for a mode 2 abort, is automatically switched to ACME by the abort sequential relays.

SYSTEM UNITS

ATTITUDE CONTROL ELECTRONICS (ACE)

The ACE package (Figure 8-4) weighs approximately 17 pounds, has a removable cover and contains ten removable module boards. These boards make up the ACE

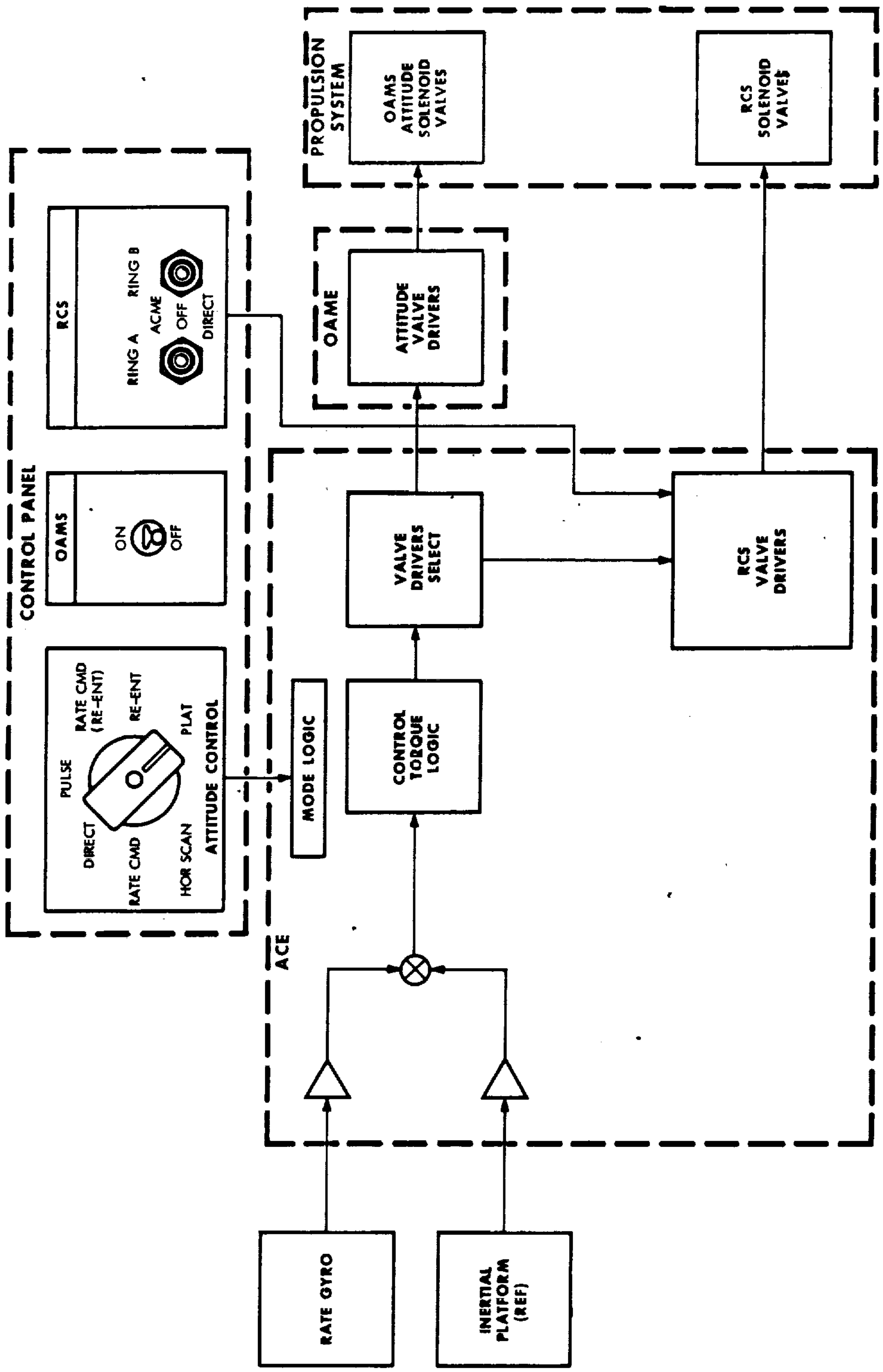


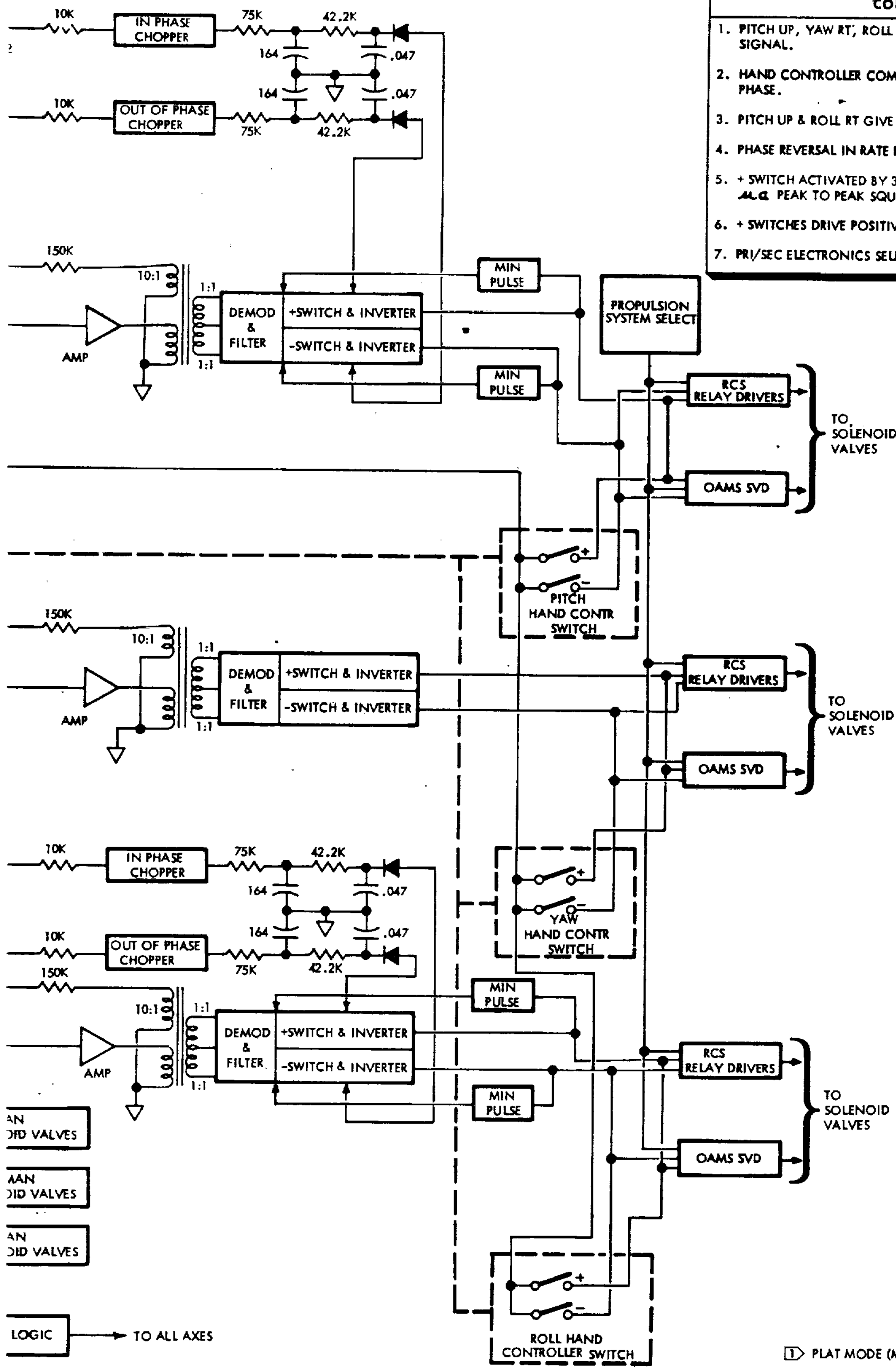
Figure 8-10 ACME Simplified Block Diagram (Platform Mode) (Spacecraft 7)

FM2-8-10

logic circuitry and consist of the following: a mode logic board, an AC signal processing board, three axis logic boards, three relay boards, a power supply board (+20, +10, -10 VDC) and a lag network board. These replaceable module boards perform the signal processing for the three axis control and convert signal inputs into an appropriate thruster firing command.

Functional Operation

Input signals to ACE are dependent on attitude requirements of the spacecraft and are used to obtain an attitude or attitude rate correction. A functional schematic of the ACE is shown in Figure 8-11 and is sectioned to show signal processing in each of the three axis channels. ACE mode logic circuits are represented by the legend blocks at the left of Figure 8-11. The selection of a mode of attitude control, initiates transistor switching in the logic circuits pertaining to that mode. The required input signal is then switched into the proper ACE channel for processing. Additional information on mode logic switching may be found in the mode selection paragraph. Proportional circuitry consists of the signal amplifier stages (attitude and rate), switch amplifiers and the demodulator/filter stages. Attitude and rate signals to each of the pitch, yaw and roll channels are AC and are amplified to operational levels by the attitude and rate amplifiers. The outputs are summed and fed to the switch amplifiers. The output of the switch amplifier is coupled to the demodulator stage where it is converted to a DC, positive or negative, analog signal. The DC signal then energizes either the positive or negative, low-hysteresis transistor switches in the control torque logic section. An 18 millisecond switch on time control is provided by the minimum pulse



CONTROL FUNCTIONS
1. PITCH UP, YAW RT, ROLL RT, GIVE IN-PHASE RATE & ATTITUDE ERROR SIGNAL.
2. HAND CONTROLLER COMMANDS PITCH DOWN, ROLL LEFT, YAW LEFT IN-PHASE.
3. PITCH UP & ROLL RT GIVE POS HORIZON SENSOR OUTPUT.
4. PHASE REVERSAL IN RATE PRE-AMP & ATTITUDE PRE-AMP.
5. + SWITCH ACTIVATED BY 3 μa RMS IN-PHASE SINE WAVE OR BY 3.9 μa PEAK TO PEAK SQUARE WAVE.
6. + SWITCHES DRIVE POSITIVE TORQUING SOLENOID VALVES.
7. PRI/SEC ELECTRONICS SELECT BY AXIS.

ATTITUDE CONTROL MODES
M_1 = DIRECT
M_2 = PULSE
M_3 = RATE CMD
M_4 = HOR SCAN
M_5 = RE-ENTRY
M_5D = RE-ENTRY RATE CMD
M_6 = PLAT ⊲1

LOGIC TABLE (SPACECRAFT 3 & 4)'
LOGIC FUNCTIONS WHERE (') DENOTES NOT GROUND
A = M_1 + M_2 PULSE + M_4 PULSE
B = M_1 + M_2 + M_4
C'_P = M_3 + M_5 + M_5D
C'_R = M_3 + M_5 + M_5D
C'_Y = M_3 + M_5 + M_5D
D'_R = M_5
I'_P = RING A + RING B + M_5 + M_5D
I'_R = RING A + RING B + M_5 + M_5D
I'_Y = RING A + RING B + M_5 + M_5D
M'_R = M_5 + M_5D
P'_P = M_5 + M_5D (RING A & RING B)
P'_R = M_5 + M_5D (RING A & RING B)
P'_Y = M_5 + M_5D (RING A & RING B)

LOGIC TABLE (SPACECRAFT 7)
LOGIC FUNCTIONS WHERE (') DENOTES NOT GROUND
A = M_1 + M_2 PULSE + M_4 PULSE
B = M_1 + M_2 + M_4
C'_P = M_3 + M_5 + M_5D + M_6
C'_R = M_3 + M_5 + M_5D + M_6
C'_Y = M_3 + M_5 + M_5D + M_6
D'_R = M_5
I'_P = RING A + RING B + M_5 + M_5D + M_6
I'_R = RING A + RING B + M_5 + M_5D + M_6
I'_Y = RING A + RING B + M_5 + M_5D + M_6
K'_P = M_6 (A_2 + A_3)
K'_R = M_6 (A_1 + A_2 + A_3)
K'_Y = M_6 (A_2 + A_3)
M'_R = M_5 + M_5D
P'_P = M_5 + M_5D + M_6 (RING A & RING B)
P'_R = M_5 + M_5D + M_6 (RING A & RING B)
P'_Y = M_5 + M_5D + M_6 (RING A & RING B)

NOTES

1. PLAT MODE (M_6) EFFECTIVE SPACECRAFT 7.
2. ALL CAPACITANCE VALUES ARE IN MICROFARADS.

FM1-8-11A

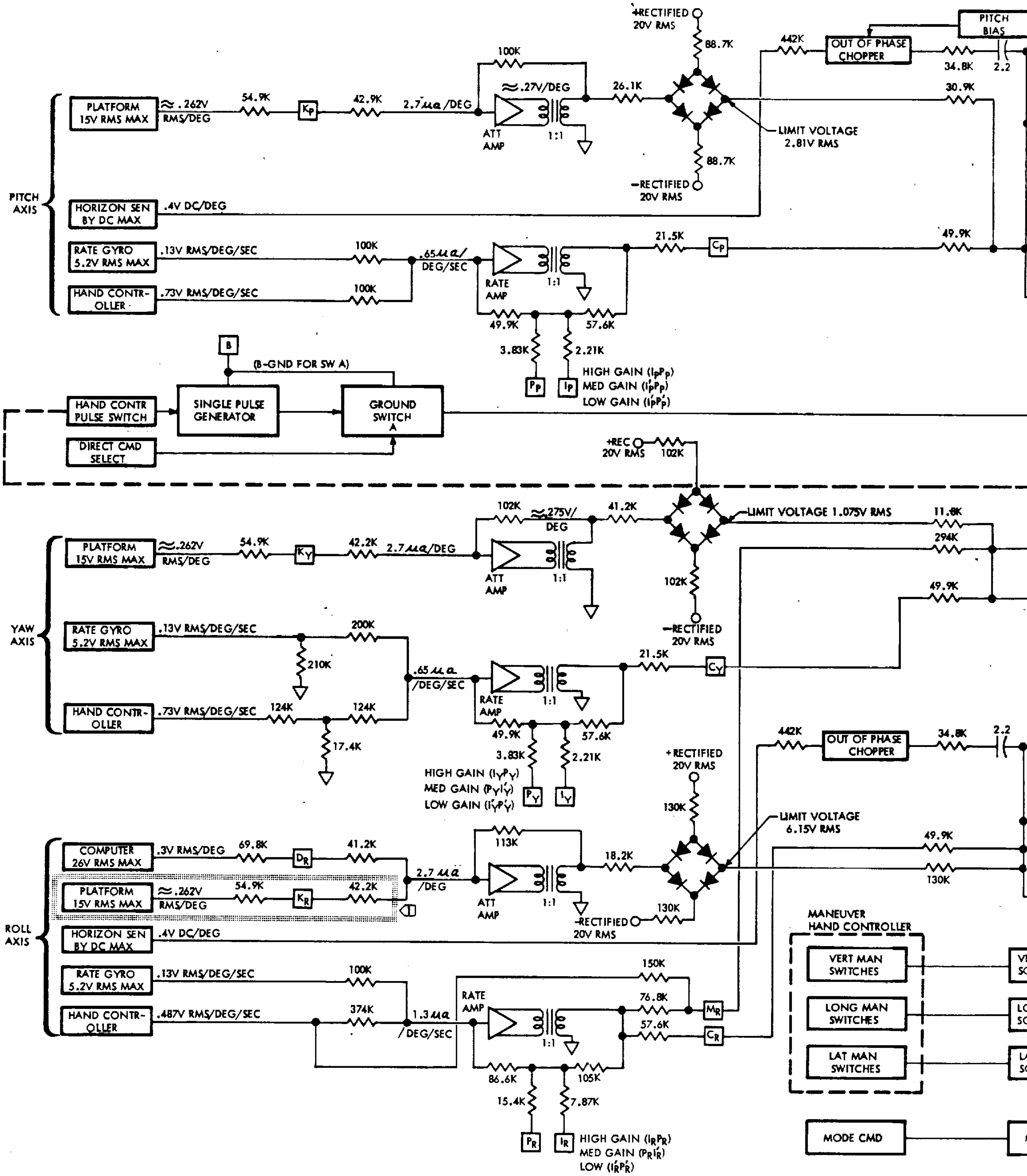
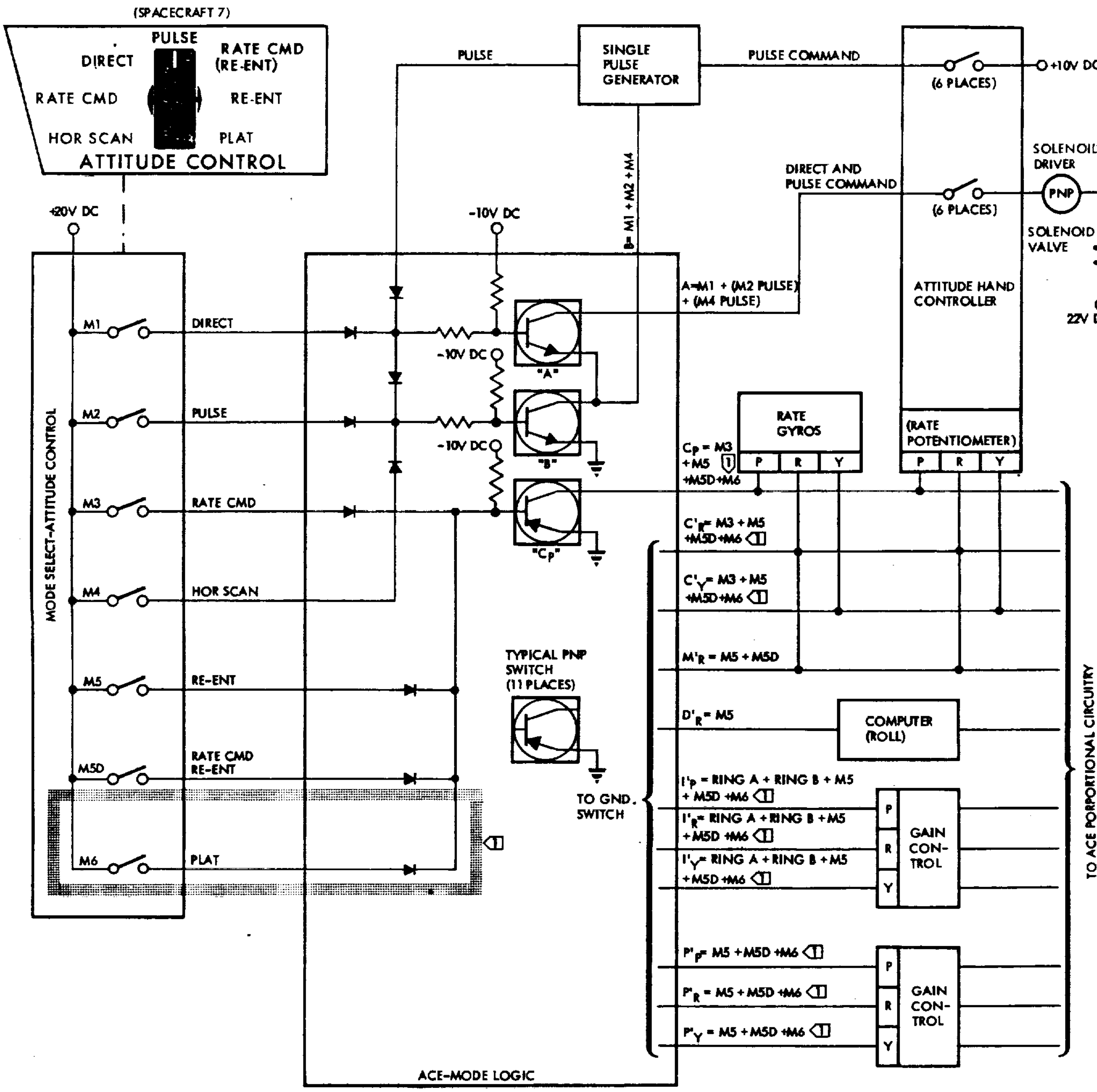


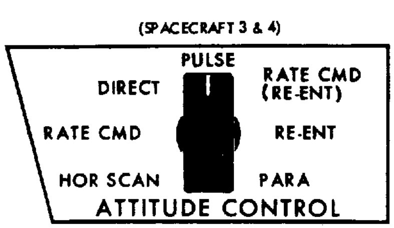
Figure 8-11 ACME-Functional Schematic

generators. Horizon sensor DC signals are chopped and amplified by the switch amplifiers, then demodulated in the same manner as AC signals. The valve driver select circuits control power and signal distribution to OAME and RCS attitude valve drivers. To turn off the OAME control system, power is applied to de-energized relays, the normally closed contacts of which complete the power and signal inputs to the OAME. Power may then be applied to the RCS ring A and/or ring B valve drivers for RCS attitude control. The ring A and ring B RCS valve drivers consists of relays, energized by transistor relay drivers.

Mode Logic Switching

Transistor switching provides the control for attitude mode signal selections, along with ACE power distribution in the horizon scan mode. These switches are represented by blocks in Figure 8-11. The logic function for each block is explained in the truth table at the right of Figure 8-11 as being ground or not ground. Figure 8-12 shows how mode control of signal selections is accomplished. The transistor switches provide a grounded or not grounded condition to attitude signals, by being in a conducting or not conducting state. Attitude reference and command signals are obtained by selecting the appropriate mode of control switch position. This applies a +20 VDC bias voltage to the base of a PNP transistor, biasing it to cut off. This ungrounded state allows the desired signal to be applied to the ACE amplifiers. The mode 1 (direct), and mode 2 (pulse), and one of the M_4 (hor scan) logic switches are NPN transistors, and conduct with the application of +20 VDC. This provides a ground circuit for hand controller commands. The pulse generator signal provides the bias voltage to turn on switch A when in the pulse or orbit modes.





NOTES

1. PLAT MODE (M6) EFFECTIVE SPACECRAFT 7.
2. IN LOGIC FUNCTIONS (') DENOTES - NOT GROUND.
3. REFER TO FIGURE 8-11 (FUNCTIONAL SCHEMATIC) FOR ACE CIRCUITRY.

Figure 8-12 ACE Mode Logic Switching-Attitude Control

FM1-8-12A

Signal Processing (See Figure 8-11)

By referring to the logic block in each channel and the mode logic table, the type signal selected for each mode of control can be determined. The P and I blocks, through mode selections, establish the gain for rate amplifier stages.

Attitude Signals

Inputs to the ACE are either in phase or out of phase AC signals (with the obvious exception of the DC horizon sensor input). A positive attitude displacement generates an in phase error signal, which in turn will command negative thrusting. A negative attitude displacement, generating an out of phase signal will command positive thrusting. By referring to the logic table, it may be seen that the selection of mode 5 provides a computer roll input through the function of logic block D_R and is the only attitude signal selected for an input to ACE. A roll attitude error or command signal is fed into the three stage attitude amplifier. The amplifier output will be used to turn on the appropriate solenoid valve driver. The bridge rectifier is used to limit attitude signal amplitude. The output of the three stage switch amplifier is transformer coupled to either the in phase or the out of phase section of the demodulator stage. The output of the demodulator stage is a full wave rectified DC signal, which is filtered and energizes either the positive or negative low hysteresis switch. Energizing the switch provides the ground for the valve drivers. The minimum pulse generator will not allow the solenoid valves to turn off in less than 18 milliseconds, thus always assuring a prescribed minimum thruster force. Minimum pulse generators are used in the pitch and roll channels only.

Rate Signals (See Figure 8-11)

Angular rate and rate command signals are provided by the logic functions of blocks Cp, Cy and Cr through the selection of modes M3, M5, M5D, and M6. Signal gains through the rate amplifiers are varied by the functions of logic blocks Ip, Iy, Ir, Pp, Py and Pr, with the selection of the re-entry modes, platform mode or RCS control. Rate signal inputs are used in the same manner as attitude signals to control solenoid valves. Roll rate signals are summed with the computer command signals and the proportional output is fed to the switch amplifiers. The function of the logic block M_R, with selection of the re-entry modes of control, provides crosscoupling of roll rates into the yaw axis for re-entry control. Roll rate signals are proportionally coupled into yaw. This provides an opposite phase signal for cancellation of part of the yaw rate command signal for proper stability.

Horizon Sensor Signals

Sensor pitch and roll signals are positive or negative DC and are fed directly to out of phase choppers in ACE. A -5 degree pitch bias voltage is summed with horizon sensor outputs for pitch down orientation. The output of the out of phase chopper will be of a phase opposite the attitude displacement. This signal is then amplified and processed by the on-off logic, in the same manner as an AC attitude signal.

The horizon scan mode in addition to circuits utilized by other modes, energizes the resistance - capacitance lag feedback networks and choppers for either the in or out of phase signal. The lag network discharge rate, along with the minimum pulse generator, provides anti-hunting control. (Hunting would result from the slow response of the horizon sensors if no anti-hunt control was used.)

RCS Valve Drivers

The RCS solenoid valve drivers (Figure 8-13) are relays with normally open contacts connected between the solenoid valve and the RCS ring switch and provides a circuit ground when the switch is in the ACME position. The relays are energized by transistor relay drivers, which conduct upon receiving thruster firing commands from the control torque logic switches or the attitude hand controller direct switches.

ORBIT ATTITUDE AND MANEUVER ELECTRONICS (OAME)

This unit (Figure 8-4) weighs approximately 8 pounds, has a removable cover and contains three removable module boards (2-relay boards and 1-component module board) and fixed mounted components. These replaceable module boards in conjunction with the fixed components function as attitude and maneuver valve drivers.

Functional Operation

Attitude Control

Attitude commands to the OAME are either positive or negative thruster firing commands to the solenoid valve drivers, from the control torque logic section of ACE. (See Figure 8-13). Upon receiving command signals, the valve driver transistors will conduct. This provides the circuit grounds to energize the solenoid valves of the propulsion system. Zener diode spike suppression is provided to limit the voltage generated when thruster power is interrupted.

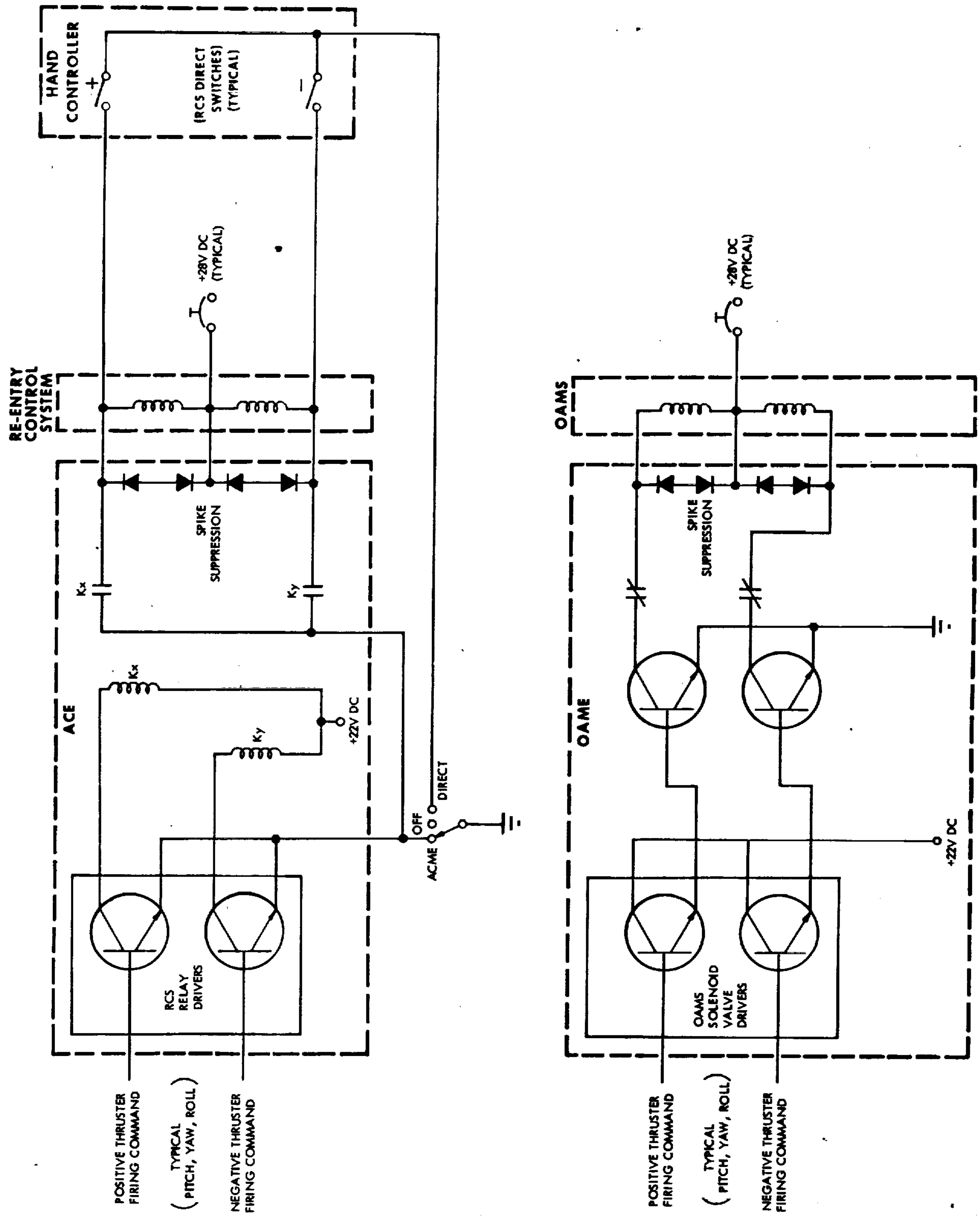


Figure 8-13 RCS & OAMS Attitude Valve Drivers

FM2-8-13

Maneuver Control

Maneuver commands to the OAME originate from either maneuver hand controller (Figure 8-14). Translational command signals are provided by applying a circuit ground through the proper hand controller switch, to the valve driver relays. Upon actuation of the relay, a normally open relay contact is closed. This applies the circuit ground to the OAMS valve solenoids for thruster firing. Conventional diode spike suppression is provided to limit the voltage spike generated when thruster power is interrupted.

RATE GYRO PACKAGE

The rate gyro package (Figure 8-4) contains three rate gyros, each individually mounted and hermetically sealed. The gyros are orthogonally mounted for rate sensing in all three axes. The rate gyro package provides AC analog outputs, proportional to mechanical rate inputs. Application of a gimbal torquer current, and monitoring the spin motor synchronization, provides a check of gyro operation and pickoff output during ground checkout. Each gyro is separately excited so that any individual gyro may be turned off, without affecting operation of the other two. Two gyro packages are provided for redundancy, and have a total weight of approximately 8 pounds.

POWER INVERTER PACKAGE

The power inverter (Figure 8-4) converts spacecraft DC power into AC power for use by the ACME sub-systems and horizon sensors. The unit weighs approximately 7 pounds and consists of the following: current and voltage regulators, oscillator, power amplifier, output filter, regulator-controller, switching

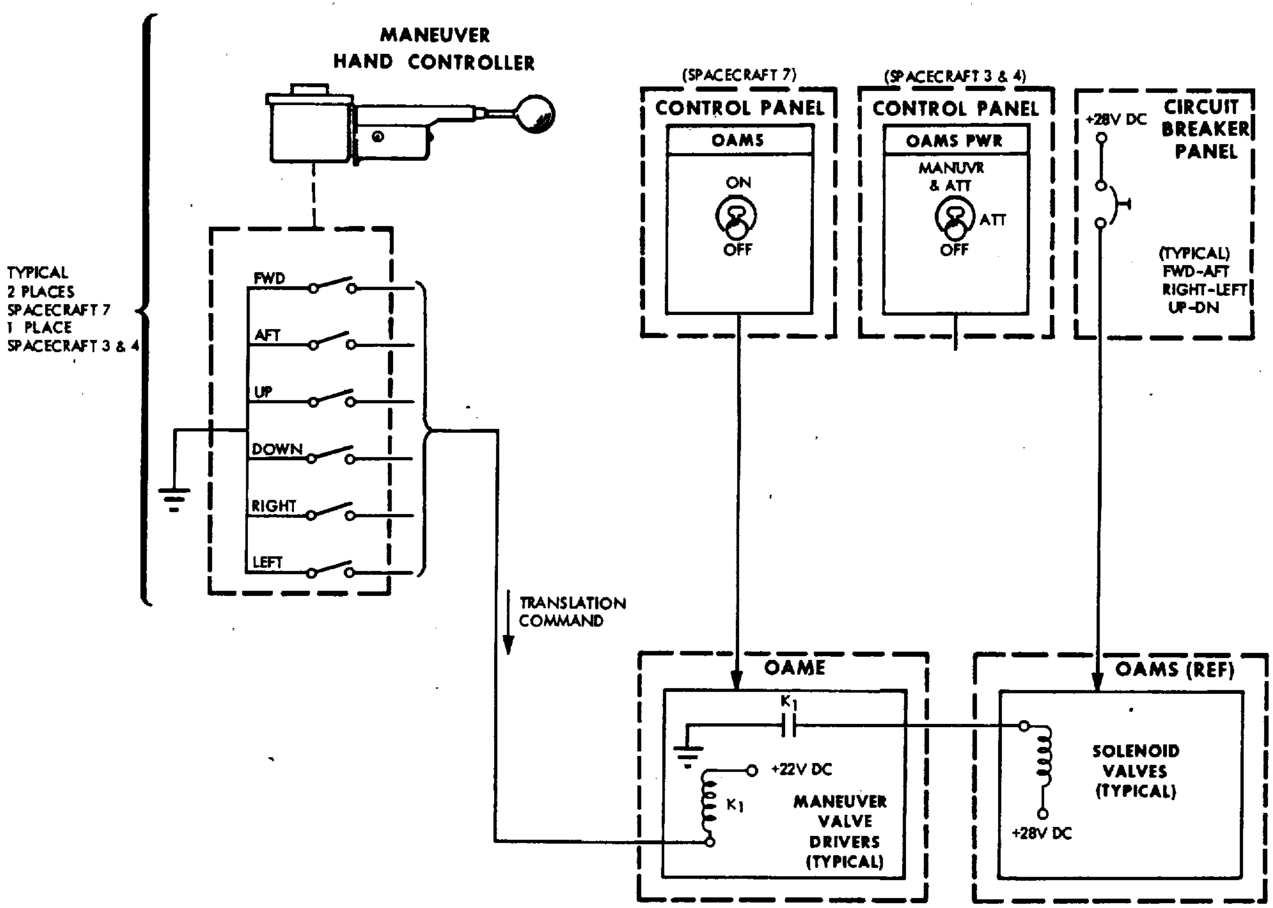


Figure 8-14 ACME Maneuver Control-Simplified Block Diagram

FM1-8-14A

regulator and oscillator starter. The 26 VAC, 400 cps power inverter output is supplied to the following:

a. ACE Power Supply: reference power for the choppers, demodulators and DC biasing voltages.

b. Rate Gyros: 20 watts starting power and 16 watts running power for motor and pickoff excitation.

c. Horizon Sensors: 11 watts operational power, as reference for bias voltages and pickoff excitation.

d. Attitude Hand Controller: 0.5 watts for potentiometer excitation.

e. Telemetry: 1.0 watts for demodulation reference.

f. FDI: 8.2 watts

INERTIAL GUIDANCE SYSTEM

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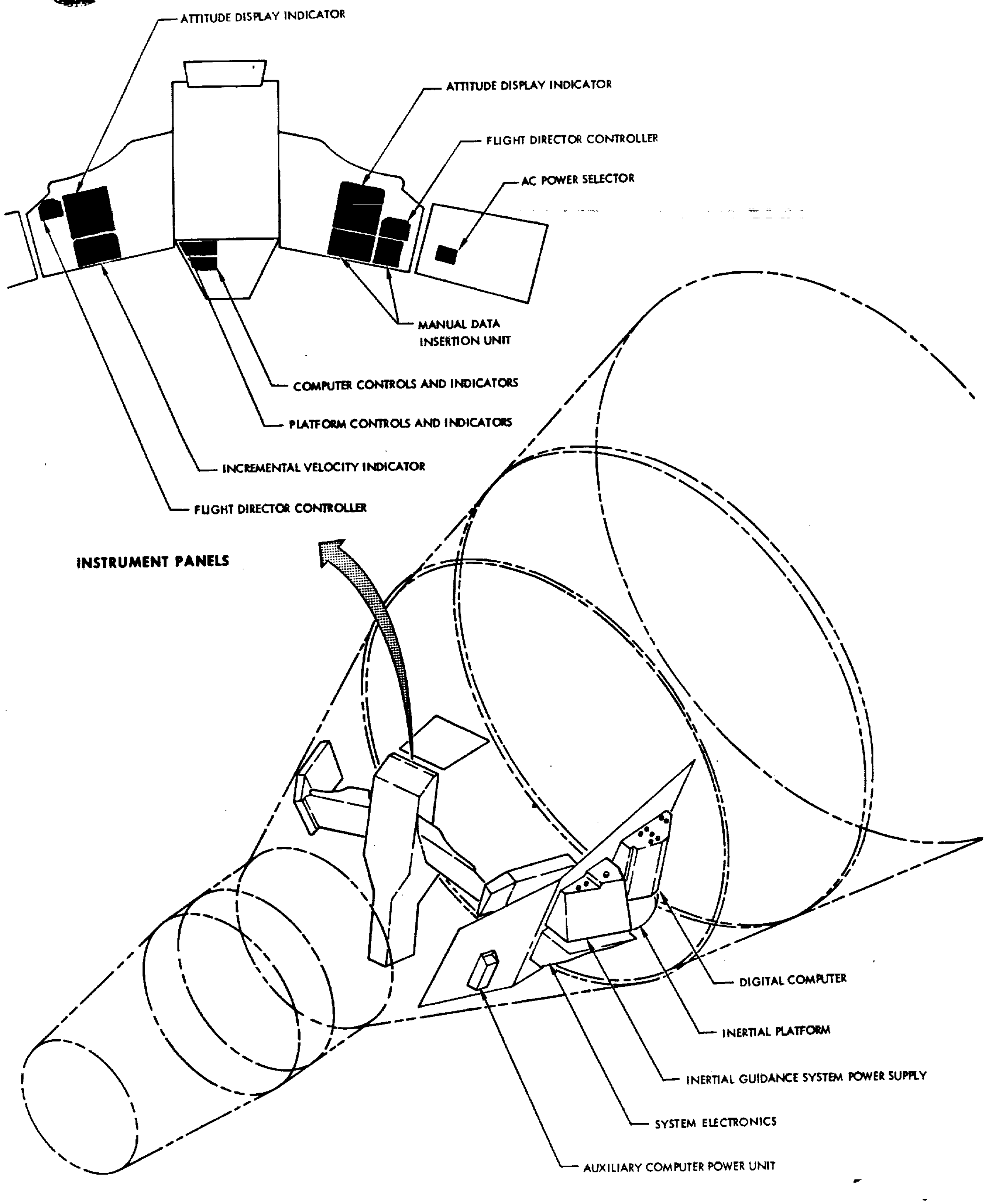

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Figure 8-15 Inertial Guidance System

INERTIAL GUIDANCE SYSTEM

SYSTEM DESCRIPTION

The Inertial Guidance System (IGS) consists of an inertial measurement unit, an auxiliary computer power unit, an on-board computer, and associated controls and indicators. The location of all IGS components is illustrated in Figure 8-15. Controls and indicators are located inside the pressurized cabin area. The inertial measurement unit, auxiliary computer power unit, and the on-board computer are located in the unpressurized left equipment bay.

INERTIAL MEASUREMENT UNIT

The Inertial Measurement Unit (IMU) consists of three separate packages: the inertial platform, system electronics, and IGS power supply. All three packages function together to provide inertial attitude and acceleration information. Attitude measurements are utilized for automatic control, computations, and visual display. Acceleration measurements are utilized for insertion, orbit correction, and retrograde computations and displays. IMU operation is controlled by a mode selector. Cage, alignment, orbit rate, and inertial modes are available. Platform attitude measurements are available to each pilot on his attitude display group. The IMU is also capable of providing 400 cps power to ACME inverter loads. An AC POWER switch allows the pilot to select the source of 400 cps ACME power.

AUXILIARY COMPUTER POWER UNIT

The Auxiliary Computer Power Unit (ACPU) provides protection, for the computer, from spacecraft bus voltage variations. If bus voltage drops momentarily, the

ACPU supplies temporary computer power. If bus voltage remains depressed, the computer is automatically turned off. The ACPU is activated by the computer power switch.

ON-BOARD COMPUTER

The On-Board Computer (OBC) provides the necessary parameter storage and computation facilities for guidance and control. Computations are utilized for insertion, orbit correction and re-entry guidance. A mode selector determines the type of computations to be performed. A START switch allows the astronaut to initiate certain computations at his discretion. The COMP light indicates the start and completion of a computation. A MALF light indicates the operational status of the computer and a RESET switch provides the capability to reset the computer in case of temporary malfunctions. A Manual Data Insertion Unit (MDIU) allows the pilot to communicate directly with the computer. Specific parameters can be inserted, read out, or cleared from the computer memory. An Incremental Velocity Indicator (IVI) displays velocity changes. Changes can be measured or computed, depending on computer mode.

SYSTEM OPERATION

Operation of the IGS is dependent on mission phase. Components of IGS are utilized from pre-launch through re-entry phases. Landing phase is not controllable and therefore no IGS functions are required. The computer and platform each have mode selectors and can perform independent functions. However, when computations are to be made concerning attitude or acceleration, the two units must be used together.

PRE-LAUNCH PHASE

Pre-launch phase consists of the last 150 minutes before launch. This phase is utilized to warm-up, check-out, program, and align IGS equipment. After warm-up, the computer performs a series of self checks to insure proper operation. Information not previously programmed but essential to the mission is now fed into the computer. AGE equipment utilizes accelerometer outputs to align IMU pitch and yaw gimbals with the local vertical. The roll gimbal is aligned to the desired launch azimuth by AGE equipment.

LAUNCH PHASE

Launch phase starts at lift-off and lasts through insertion. During the first and second stage boost portion of launch, the guidance functions are performed by the booster autopilot. If the primary booster guidance system should fail, a Malfunction Detection System (MDS) provides automatic switchover to back-up (Gemini) guidance. Back-up ascent guidance can also be selected manually, at the discretion of the command pilot. The computer has been programmed with launch parameters and the IMU provides continuous inertial reference for back-up ascent guidance. To minimize launch errors, the computer is updated by ground stations throughout the launch phase. In the back-up ascent guidance operation, the computer provides steering and booster cut-off commands to the secondary booster autopilot. The computer also supplies attitude error signals to the flight director needles. The IMU provides inertial attitude reference to the attitude ball. At Second Stage Engine Cut-Off (SSECO), guidance control is switched from booster to Gemini IGS. The computer starts insertion computations at SSECO and, at spacecraft separation, displays the incremental velocity change required for insertion in the desired orbit. When the required velocity change appears, the command pilot will accelerate the spacecraft to insertion velocity. During acceleration, the IMU supplies attitude and velocity changes to the computer. The computer continuously subtracts measured acceleration from required acceleration on the display. When insertion has been achieved, the incremental velocity indication will be zero along all three axes.

ORBIT PHASE

Orbit phase consists of that time between insertion and the start of retrograde sequence. If the IGS is not to be used for long periods of time, it can be turned off to conserve power. If the platform has been turned off, it should be warmed up in the CAGE mode approximately one hour before critical alignment. The computer should be turned on in the pre-launch mode and allowed 20 seconds for self checks before changing modes. IGS operation, during orbit, is divided into three separate operations. The initial part of orbit is used for check out and alignment. The major part of orbit is used to perform experiments and orbital maneuvers and the final portion is used in preparation for retrograde and re-entry.

Check-Out & Alignment

Immediately after orbit confirmation the spacecraft is maneuvered to small end forward and the platform aligned with the horizon sensors. Horizon sensor outputs are used to align pitch and roll gimbals in the platform. The yaw gimbal is aligned through gyrocompassing techniques using the roll gyro output. This output is used to align the yaw gyro to the orbit plane. The platform alignment will be maintained by the horizon sensors as long as SEF or BEF modes are used. ORB RATE mode is used when maneuvers are to be performed. ORB RATE is an inertially free mode except for the pitch gyro which is torqued at approximately four degrees per minute (orbit rate). The purpose of torquing the pitch gyro is to maintain a horizontal attitude with respect to the earth. If ORB RATE mode is used for long periods of time, drift errors can occur. To eliminate errors due to gyro drift, the mode is switched back to SEF or BEF for automatic alignment.

Orbital Maneuvers

IGS operation during orbital maneuvers consists of performing inertial measurements and maneuver computations. Platform alignment is performed in SEF or BEF mode prior to initiating a maneuver. The computer START button is pressed to initiate computation of velocity changes and computed velocity requirements are automatically displayed on the IVI. Flight director needles are referenced to the computer during maneuvers and indicate the attitude in which translational thrust should be applied. When the spacecraft is in the correct attitude for a maneuver, all of the incremental velocity indication will be along the forward-aft translational axis. As thrust is applied, the IMU supplies the computer with attitude and acceleration information to continuously update the IVI indications. When the maneuver has been completed, the platform can be realigned to the horizon sensors.

Preparation for Retrograde & Re-Entry

Preparation for retrograde and re-entry is performed in the last hour before retrograde sequence. If the IMU has been turned off, it must be turned on one hour before retrograde. (The gyros and accelerometers require approximately one half hour to warm up and another half hour is required for stabilization and alignment.) The attitude ball will indicate when platform gimbals are aligned to spacecraft axes. At this time, the spacecraft is maneuvered to Blunt End Forward (BEF) and the platform aligned with the horizon sensors. The platform remains in BEF mode to maintain alignment until retrograde sequence. The computer retrograde initial conditions are checked and if necessary updated by either ground tracking stations or the pilot. Preparation for retrograde and re-entry is completed by placing the computer in RNTY mode.

RETROGRADE PHASE

Retrograde phase starts at five minutes prior to retrofire on spacecraft 3 and 4 (256 seconds prior to retrofire on spacecraft 7) and ends approximately twenty-five seconds after retrofire initiation. At the start of retrograde phase, a minus sixteen degree bias is placed on the pitch needle of the attitude indicator. At time-to-go to retrograde minus 30 seconds (T_R-30 seconds), the platform is placed in ORB RATE mode. While the retro-rockets are firing (approximately 22 seconds), the acceleration and attitude are monitored by the IMU and supplied to the computer for use in re-entry computations. The computer starts computations for re-entry at retrofire. Computations are based on the time of retrofire, inertial position and attitude, and retrograde acceleration.

RE-ENTRY PHASE

Re-entry phase starts immediately after the retro rockets stop firing and lasts until drogue chute deployment. After retrograde, a 180° roll maneuver is performed and pitch attitude is adjusted so that the horizon can be used as a visual attitude reference. The spacecraft attitude is controlled by visual observation of the horizon until the computer commands a re-entry attitude at approximately 400,000 feet. The spacecraft is then controlled to null the flight director needles. Flight director needles are referenced to the computer during re-entry. The IMU supplies inertial attitude and acceleration signals to the computer. Bank angle commands are computed and displayed on the roll needle for down range and cross range error correction. The bank angle commands last between 0 and 500 seconds depending on the amount of down range and cross range error. Pitch and yaw needles display down range and cross range errors respectively. Upon completion of the bank angle commands (spacecraft on target), a roll rate of 15 degrees per second is commanded by the computer. At approximately 80,000 feet, the computer commands an attitude suitable for drogue chute deployment. Immediately after drogue deployment the IGS equipment is turned off.

CONTROLS AND INDICATORS

Attitude Display Group

The Attitude Display Group (ADG) (Figure 8-16) consists of a Flight Director Indicator (FDI) a Flight Director Controller (FDC) and their associated amplifiers. Three types of displays (attitude, attitude rate, and ADG power off) are provided by the FDI. A three axis sphere with 360 degrees of freedom in each axis continuously displays attitude information. The sphere is slaved to the inertial platform gimbals and always indicates platform attitude. Three needle type indicators display attitude and/or attitude rate information as selected by the pilot. Information displayed on the needles is provided by the computer, platform and rate gyros. A scale selector is included in the

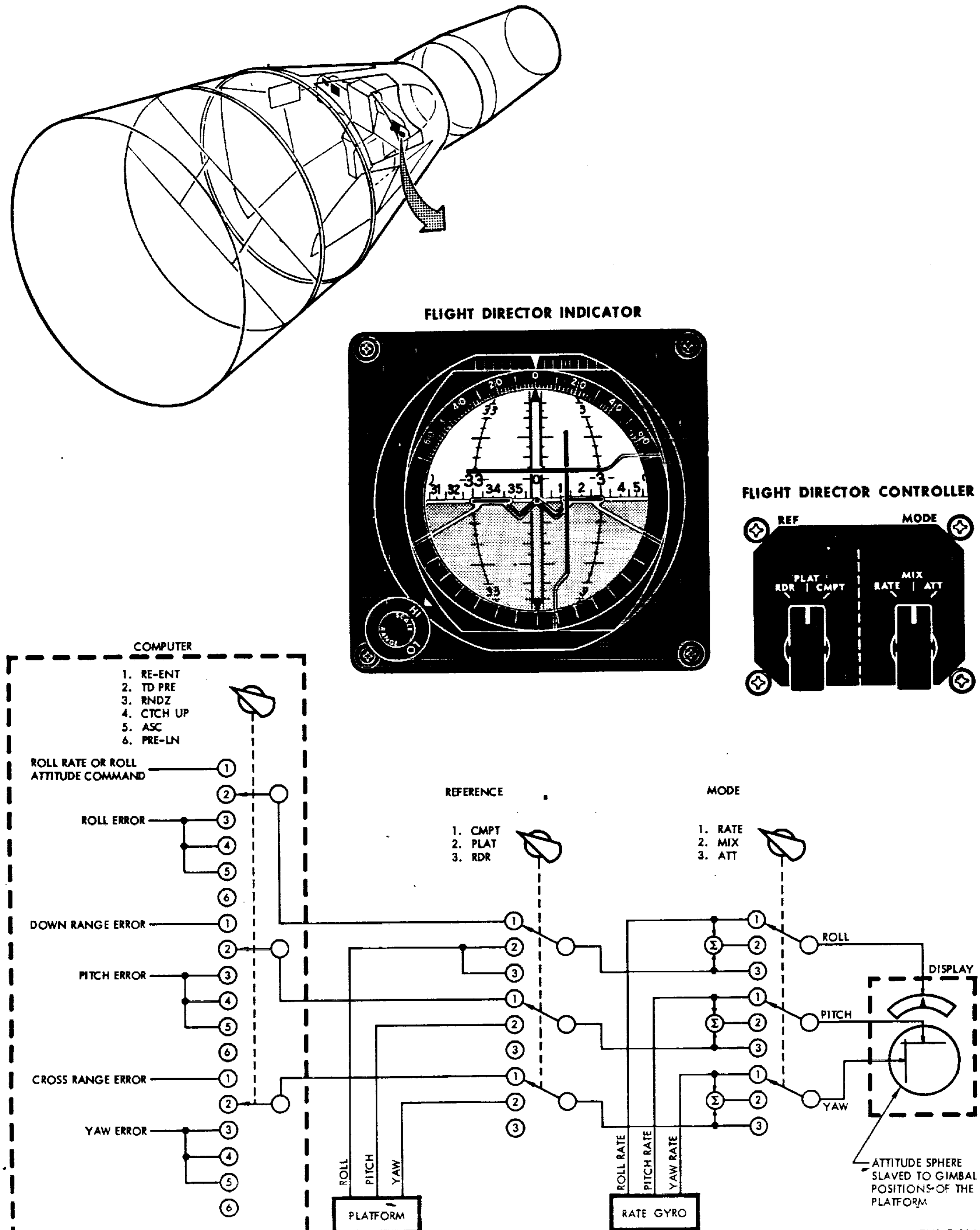

Figure 8-16 Attitude Display Group

FDI to allow the selection of HI or LO scale indications on the needles. The FDC is used to select the source and type of display on the needles. Figure 8-16 includes a simplified schematic of the FDC switching and indicates the source and type of signal available. Since the computer is capable of producing different types of signals, the computer mode selector is included in the schematic. The FDC reference selector determines the source of display information. The FDC mode selector determines the type of signal displayed.

Manual Data Insertion Unit

The Manual Data Insertion Unit (MDIU) consists of a ten digit keyboard and a seven digit register. The MDIU allows the pilot to communicate directly with the on-board computer. Provision is made to enter, cancel or read out information. The keyboard is used to address a specific location in the computer and set up coded messages for insertion. The first two keys that are pressed address the computer memory word location and the next five set up a coded message. Keys are pressed in a "most significant bit first" order. Negative values are inserted by making the first number of the message a 9. The 9 then represents a minus sign and not a number. The seven digit register is used to monitor addresses and messages entered into or read out of the computer. Push button switches are included on the register panel to READ OUT, CLEAR, and ENTER the messages. Information can also be inserted in the computer by the ground tracking stations which have digital command system capabilities.

Incremental Velocity Indicator

The Incremental Velocity Indicator (IVI) provides a display of computed velocity increments required for, or resulting from, a specific maneuver. The IVI is controlled through the on-board computer. Displays are utilized for orbit insertion, orbit correction and retrograde. Velocity increments are provided along each of the spacecraft translational axis. Controls are included to manually insert plus or minus velocity increments into the IVI.

Computer Controls

Computer controls consist of: a COMPUTER mode selector, a START switch, a COMP light, a MALF light, a RESET switch, and an ON-OFF switch. The COMPUTER mode selector is a rotary switch which selects the type of computations to be performed. Modes of operation correspond to the mission phase in which they are utilized. The COMP light indicates when the computer is running through its program and provides a means of checking computer sequencing. The START switch is utilized for manual initiation of certain computations.

NOTE

The START switch must be operated in conjunction with the computer mode selector and the COMP light.

The MALF light indicates when a malfunction has occurred and the RESET switch resets the computer malfunction indicator. The RESET switch is only capable of resetting the computer for momentary malfunctions. An ON-OFF switch controls power to the computer and the auxiliary computer power unit.

IMU Controls & Indicators

The IMU controls and indicators consist of: a PLATFORM mode selector, an ACC light, an ATT light, a RESET switch, and an AC POWER selector. The PLATFORM mode selector is a seven position rotary switch which, in conjunction with the AC POWER selector, turns the platform on and off as well as control the mode of operation. Two cage modes, two align modes, one free mode, and an orbit rate mode of operation are selectable. The align modes are SEF and BEF. The ACC light indicates when a malfunction has occurred in the accelerometer portion of the IMU. The ATT light indicates when a malfunction has occurred in the attitude portion of the IMU. The RESET switch will turn off the lights, indicating that the IMU has returned to normal operation. The RESET switch works for momentary malfunctions of either type. Inability to reset the lights indicates a permanent malfunction. The AC POWER selector allows the pilot to turn the IGS inverter on without operating the platform or electronics circuits.

SYSTEM UNITS

INERTIAL MEASUREMENT UNIT

The Inertial Measurement Unit (IMU) is the inertial attitude and acceleration reference for the Gemini spacecraft. The IMU consists of three separate packages: the inertial platform, platform electronics, and IGS power supply. All three packages conform to spacecraft contours for mounting convenience and have a total weight of 130 pounds. A functional block diagram (Figure 8-17) indicates functions and signal routing throughout all three packages. In addition to attitude and acceleration reference, the IMU provides AC and DC power for use

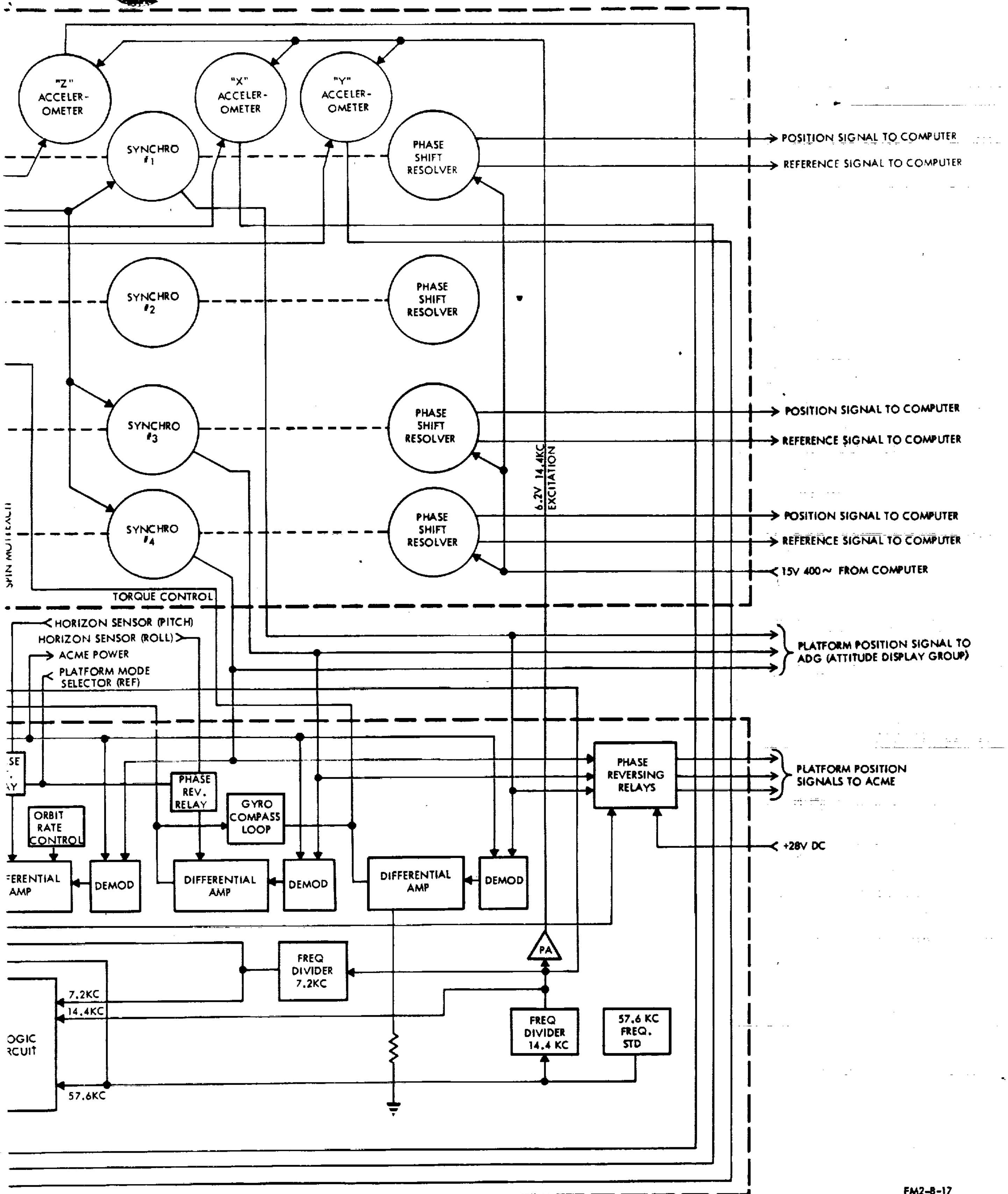

Figure 8-17 IMU Functional Block Diagram

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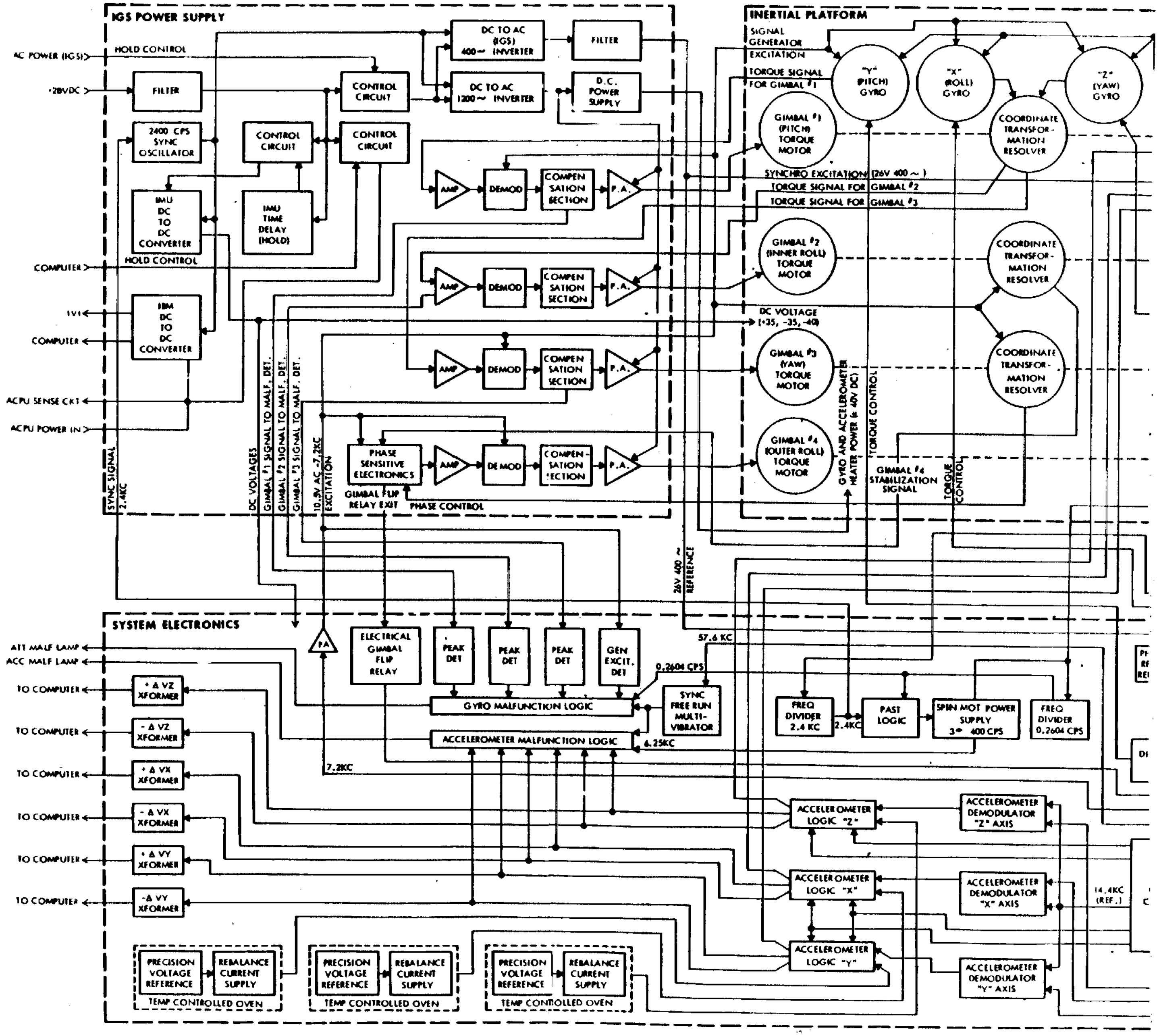

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in other units of guidance and control. The platform and electronics packages are mounted on cold plates to prevent overheating.

NOTE

References to x, y, and z attitude and translational axes pertain to inertial guidance only and should not be confused with structural coordinate axes.

Inertial Platform

The inertial platform (Figure 8-18) is a four gimbal assembly containing three miniature integrating gyros and three pendulous accelerometers. Gimbals allow the gyro mounting frame (pitch block) to remain in a fixed attitude while the housing moves freely about them. Major components of the platform are: a housing, gimbal structure, torque motors, gimbal angle synchros, resolvers, gyros and accelerometers. The gimbals from inside to outside are: pitch, inner roll, yaw and outer roll. All gimbals, except inner roll, have 360 degrees of freedom. The inner roll gimbal is limited to plus and minus 15 degrees. Two roll gimbals are used to eliminate the possibility of gimbal lock. Gimbal lock can occur on a three gimbal structure when an attitude of 0 degrees yaw, 0 degrees pitch, and 90 degrees roll exists. At this time, the roll and yaw gimbals are in the same plane and the yaw gimbal cannot move about its axis (gimbal lock). In the four gimbal platform, an angle of 90 degrees is

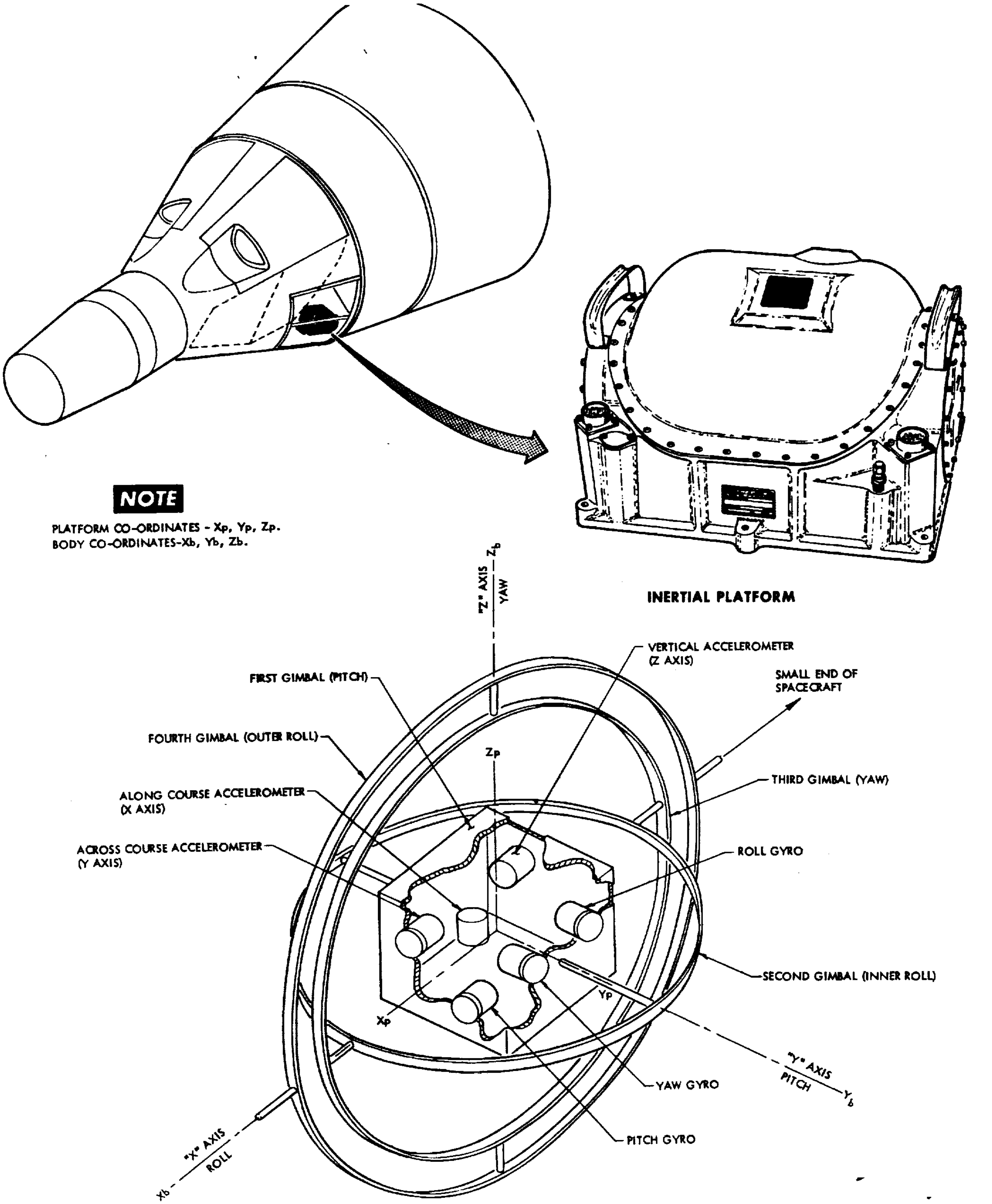

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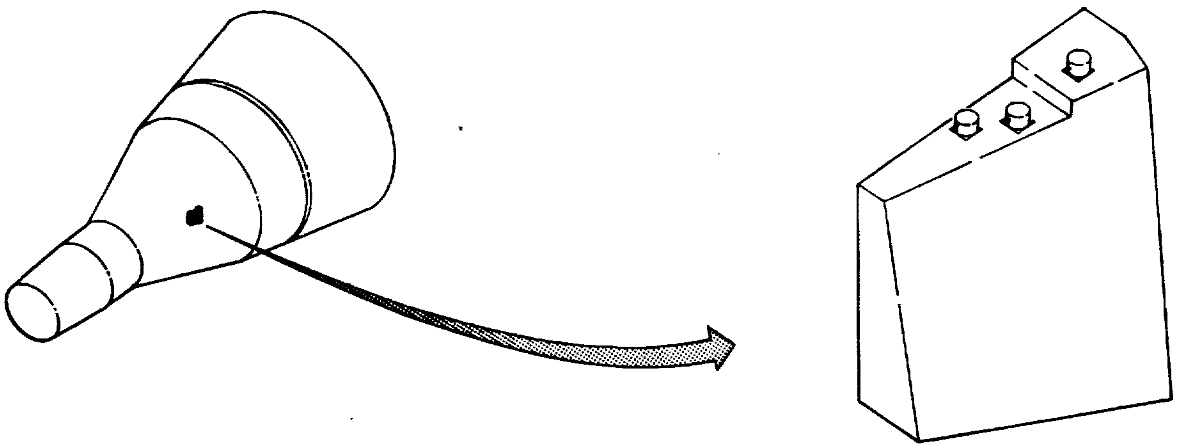
Figure 8-18 Inertial Platform Gimbal Structure maintained between the inner roll and yaw gimbals thus preventing gimbal lock. The inertial components are mounted in the innermost gimbal casting (pitch block) for rigidity and shielding from thermal effects. The gyros and associated servo loops maintain the pitch block in a fixed relationship with the reference coordinate system. The accelerometer input axes are aligned with the three mutually perpendicular axes of the pitch block. Two sealed optical quality windows are provided in the housing for alignment and testing. Both windows provide optical access to an alignment cube located on the stable element.

System Electronics

The system electronics package contains the circuitry necessary for operation of the IMU. Circuits are provided for gyro torque control, timing logic, spin motor power, accererometer logic, accelerometer rebalance, and malfunction detection. Relays provide remote mode control of the above circuits.

IGS Power Supply

The IGS power supply (Figure 8-19) contains gimbal control electronics and the static power supply unit. Gimbal control electronics drive torque motors in the platform. Separate control circuits are provided for each gimbal. The static power supply provides the electrical power for the IMU, OBC, ACPU, MDIU, IVI, ACME, and horizon sensors. Figure 8-19 indicates the types of power available and the units to which they are supplied.



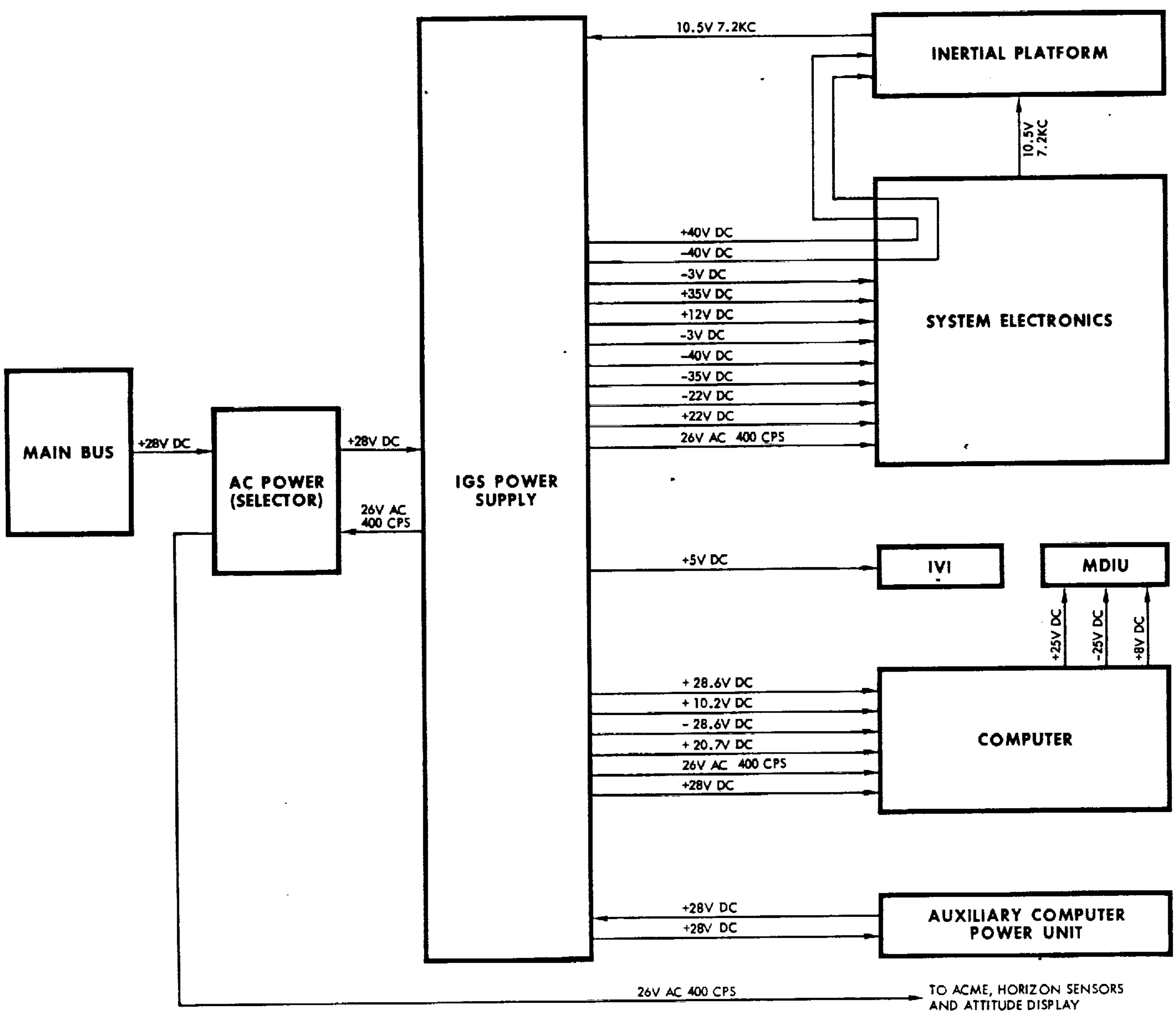


Figure 8-19 IGS Power Supply

Attitude Measurement

Attitude measurements are made from inertial platform gimbals and reflect the difference between spacecraft and gimbal attitudes. Platform gimbals are maintained in essentially a fixed inertial attitude by gimbal control electronics. As the spacecraft moves about the attitude axes, friction transfers some of the movement to platform gimbals. Three miniature gyros are used to sense minute gimbal attitude changes. When gyros sense a change in attitude, they produce a signal proportional to the attitude error. Gyro outputs are then used by gimbal control circuits to drive gimbals to their original inertial attitude. Gimbal positions, relative to the spacecraft, are measured by synchros and resolvers. Synchro outputs are provided for attitude display, automatic attitude control, and gyro alignment. Two types of resolvers, phase shift and coordinate transformation, are used. Phase shift resolvers provide gimbal angle information to the computer. Coordinate transformation resolvers provide attitude signals for gimbal control purposes.

Modes of Operation

Seven modes of operation are selectable by the pilot. The modes, in order of switch position are: OFF, CAGE, SEF, ORB RATE, BEF, CAGE, and FREE. The CAGE position is used for IMU warm-up and to align the platform gimbals with spacecraft body axes. Platform gimbals are caged prior to fine alignment with the horizon sensors. In the cage mode, gimbals are torqued by synchro outputs until a null is obtained on the synchro. When synchro outputs reach null, torquing stops and the gimbals are aligned with spacecraft axes. SEF (small end forward) mode is used to align the platform with the horizon sensors when the spacecraft is flying small end forward. Horizon sensor pitch and roll outputs are compared with synchro outputs and the difference used to torque gimbals. When synchro and horizon sensor outputs are balanced, the gimbals are aligned to earth local vertical. A gyro compass loop aligns the yaw gimbal with the orbit plane.

NOTE

If horizon sensors lose track during either SEF or BEF alignment modes, the platform is automatically switched to orbit rate mode.

ORB RATE (orbit rate) mode is used to maintain attitude reference during spacecraft maneuvers. Orbit rate mode is inertially free except for the pitch gyro. The pitch gyro is torqued at approximately four degrees per minute to maintain a horizontal attitude with respect to the earth. If orbit rate mode is used for long periods of time, drift can cause excessive errors in the platform. BEF (blunt end forward) mode is the same as SEF except that relays reverse the phase of horizon sensor inputs. The second CAGE mode allows the platform to be caged in blunt end forward without switching back through other modes. FREE mode is used during launch and re-entry phases. Free mode is completely inertial and the only torquing employed is for drift compensation.

NOTE

Free mode is selected automatically by the Sequential System at retrofire.

Gimbal Control Circuits

Four separate servo loops provide gimbal attitude control. Figure 8-17 illustrates the signal flow through all four loops. Gyro signal generator outputs are used either directly or through resolvers as the reference for gimbal control. Both phase and amplitude of signal generator outputs are functions of gimbal attitude. Gimbal number one (pitch) is controlled directly by the pitch gyro output. Error signals produced by the pitch gyro are amplified, demodulated, and compensated, then used to drive the pitch gimbal torque motor. The first amplifier raises the signal to the level suitable for demodulation. After amplification, the signal is demodulated to remove the 7.2 KC carrier. A compensation section keeps the signal within the rate characteristics necessary for loop stability. When the signal is properly conditioned by the compensation section, it goes to a power amplifier. The power amplifier supplies the current required to drive gimbal torque motors. Torque motors then drive gimbals maintaining gyro outputs at, or very near, null.

Roll and yaw servo loops utilize resolvers to correlate gimbal angles with gyro outputs. Inner roll and yaw gimbals are controlled by a coordinate transformation resolver mounted on the pitch gimbal. When the spacecraft is at any pitch attitude other than 0 or 180 degrees, some roll motion is sensed by the yaw gyro and some yaw motion is sensed by the roll gyro. The amount of roll motion sensed by the yaw gyro is proportional to the pitch gimbal angle. The resolver, mounted on the pitch gimbal, coordinates roll and yaw gyro output with pitch gimbal angle. Resolver output is then conditioned in the same manner as in the pitch servo loop to drive inner roll and yaw gimbals.

The outer roll gimbal is servo driven from the inner roll gimbal resolver. A coordinate transformation resolver, mounted on the inner roll gimbal, monitors the angle between inner roll and yaw gimbals. If the angle is anything other than 90 degrees, an error signal is produced by the resolver. The error signal is conditioned in the same manner as in the pitch servo loop to drive the outer roll gimbal. One additional circuit (phase sensitive electronics) is included in the outer roll servo loop. The outer roll gimbal torque motor is mounted on the platform housing and moves about the stable element with the spacecraft. As the spacecraft moves through 90 degrees in yaw, the direction that the outer roll gimbal torque motor must rotate, to compensate for spacecraft roll, reverses. Phase sensitive electronics and a resolver provide the phase reversal necessary for control. The resolver is used to measure rotation of the yaw gimbal about the yaw axis. As the gimbal rotates through 90 degrees in yaw, the resolver output changes phase. Resolver output is compared to a reference phase by the phase sensitive electronics. When the resolver output changes phase, the torque motor drive signal is reversed.

Pre-Launch Alignment

The IMU is the inertial reference for back-up ascent guidance and must, therefore, be aligned for that purpose. The platform is aligned to local vertical and the launch azimuth. Platform X and Y accelerometers are the reference for local vertical alignment. When the platform is aligned to the local vertical, X and Y accelerometers are level and cannot sense any acceleration due to gravity. If any acceleration is sensed, the platform is not properly aligned and must be torqued until no error signal exists. The accelerometer output is used by AGE equipment to generate torque signals for the gyros. When the gyro is torqued, it produces an error signal which is used to align the gimbal. The outer roll gimbal synchro output is compared with a signal representing the launch azimuth by AGE equipment. The error signal is conditioned by AGE equipment and applied to the yaw gyro torque generator. The yaw gyro signal generator then produces a signal proportional to the input torque. Gyro output is coordinated by a resolver mounted on the pitch gimbal. Since the spacecraft is in a 90 degree pitch up attitude, essentially all of the yaw gyro output is transferred to roll gimbal control electronics. The electronics drive the roll gimbals until no error exists between synchro output and the AGE reference signal. When no error signal exists, the platform is aligned to the launch azimuth.

Orbit Alignment

Alignment of the platform in orbit is accomplished by referencing it to the horizon sensors. Placing the platform mode selector in SEF or BEF position will reference it to the horizon sensors. Pitch and roll horizon sensor outputs are compared with platform pitch and outer roll synchro outputs. Differential amplifiers produce torque control signals, proportional to the difference between sensor and synchro outputs. Torque control signals are used to drive pitch and roll gyro torque generators. Gyro signal generator outputs are then used by gimbal control electronics to drive platform gimbals. When synchro and horizon sensor outputs balance, the pitch and roll gimbals are aligned to the local vertical. The yaw gimbal is aligned to the orbit plane through a gyro compass loop. If yaw errors exist, the roll gyro will sense a component of orbit rate. The orbit rate component in the roll gyro output is used, through a gyro compass loop, to torque the yaw gyro. Yaw gyro output is then used by gimbal control electronics to drive the yaw gimbal. When the roll gyro no longer senses a component of orbit rate, the yaw gimbal is aligned to the orbit plane. All three gimbals are now aligned and will remain aligned as long as SEF or BEF modes are used. The pitch gyro will be continuously torqued (at the orbit rate) to maintain a horizontal attitude.

NOTE

If horizon sensors lose track while the platform is in SEF or BEF modes, the platform is automatically switched to orbit rate mode.

Orbit Rate Circuit

The orbit rate circuit is used to maintain alignment to the local vertical during orbit maneuvers. Local vertical cannot be provided by horizon sensors during maneuvers because they will lose track. To maintain a horizontal attitude with no external reference, the pitch gyro is torqued at approximately four degrees per minute. The torque represents the spacecraft orbit rate. Torque is obtained by placing a DC bias on the output of the pitch differential amplifier. The bias drives the pitch gyro torquer at the orbit rate. Orbit rate bias is adjustable and can be set to match orbits of various altitudes.

Phase Angle Shift Technique

Phase Angle Shift Technique (PAST) is a method of improving gyro drift repeatability. One of the factors which affects gyro drift is spin motor rotor unbalance. The effect of unbalance will vary with changes in the point of lock on with the synchronous motor's rotating field. The spin motor can lock on to a different point each time it is started. Drift errors, due to rotor unbalance, are in the order of 0.5 degrees per hour. PAST provides a means of reducing drift errors by a factor of ten. To cancel drift errors, PAST shifts the phase of spin motor excitation 30 degrees at regular intervals. Shifting the phase causes the rotor to lock on a different point each time the phase is shifted. Drifts now tend to cancel and become predictable. (When drift is predictable, it can be compensated for.) All three gyro torque control loops contain drift compensation circuits. The drift compensation circuits apply a DC bias to each gyro torque generator. Drift compensation torques the gyro in the opposite direction as predictable drift, maintaining a stable attitude.

Attitude Malfunction Detection

An attitude malfunction detection circuit performs self checks of gyro signal generator excitation, gimbal control signals, logic timing signals, and critical voltages. Gyro signal generator excitation is checked for presence and proper amplitude. Gimbal control signals are checked for the length of time signals are present. The logic timing signal (28.8 KC) is checked for presence. Critical voltages (+22V DC, -3V DC, +12V DC) are checked for presence. If a malfunction is detected, an ATT light on the control panel is automatically illuminated. If momentary malfunctions occur, the ATT indicator can be restored to normal operation by pressing the RESET button.

NOTE

If the attitude measurement circuits malfunction, the acceleration indications are not reliable. Accelerometer axes will not be properly aligned and indications are along unknown axes.

Acceleration Measurement

Acceleration is measured along three mutually perpendicular axes of the inertial platform. Sensing devices are three miniature pendulous accelerometers. The accelerometers are mounted in the platform pitch block and measure acceleration along gyro x, y, and z axes. Accelerometer signal generators produce signals whose phase is a function of the direction of acceleration. Signal generator output is used to control torque rebalance pulses. The torque rebalance pulses drive accelerometer pendulums toward their null position. Rebalance pulses are DC current whose polarity is controlled by signal generator output. The polarity of rebalance pulses indicates the direction of acceleration and the algebraic sum of the pulses indicates the amount of acceleration. Rebalance pulses are supplied to the spacecraft digital computer where they are used for computations and incremental velocity displays.

Torque Rebalance Loop

Three electrically identical torque rebalance loops are used to control accelerometer pendulum positions. Normally an analog loop would be used for this purpose; however, if an analog loop were used, the output would have to be converted to digital form for use in the computer. To eliminate the need for an analog to digital converter, a pulse rebalance loop is used. Short duration 184 milliampere DC current pulses drive the accelerometer pendulum in one direction until it passes through null. Pulses are applied at the rate of 3.6 KC. When the pendulum passes through null, signal generator output changes phase. The signal generator output is demodulated to determine the direction of the pendulum from null. Demodulator output is used by logic circuits to control the polarity of rebalance pulses. If acceleration is being sensed, there will be more pulses of one polarity than the other. If no acceleration is being sensed, the number of pulses of each polarity will be equal. In addition to controlling the polarity of rebalance pulses, logic circuits set up precision timing of the pulses. Precision frequency inputs from the timing circuits are the basis for rebalance pulse timing. Precise timing is essential because the amount of pendulum torque depends on the length of the current pulse. All pulses are precisely the same duration and amplitude, therefore total torque is dependent only on the algebraic sum of the applied pulses. Each time a rebalance pulse is applied to the accelerometer torquer, a pulse is also provided to the computer. Algebraic summation of the rebalance pulses is performed by the computer.

Pulse Rebalance Current Supply

A pulse rebalance current supply provides the required current for torque rebalance. Since acceleration measurements are based on the number of torque pulses it is essential that all pulses be as near identical as possible. To maintain a stable current, a negative feedback circuit is employed. The supply output is passed through a precision resistor and the voltage drop across the resistor is compared to a precision voltage reference. Errors detected by the comparison are used in the feedback circuit to correct any deviations in current. To further enhance stability, both the current supply and the precision voltage reference are housed in a temperature controlled oven.

Accelerometer Dither

A pendulous accelerometer, unlike a gyro, has an inherent mass unbalance. The mass unbalance is necessary to obtain the pendulum action. Due to the unbalance, perfect flotation of the pendulous gimbal cannot be achieved and consequently pressure is present on the gimbal bearing. To minimize the stiction effect, caused by bearing friction, a low amplitude oscillation is imposed on the gimbal. The oscillation (dither) prevents the gimbal from resting on its bearing long enough to cause stiction. To obtain gimbal oscillation, two signals are required: a 100 cps dither signal and a DC field current. The DC field current is superimposed on the signal generator excitation and creates a magnetic field around the gimbal. The 100 cps dither is applied to a separate (modulator) coil. The dither signal beats against the DC field, causing the gimbal to oscillate up and down. The dither motion is not around the output axis and consequently no motion is sensed by the signal generator.

Accelerometer Malfunction Detection

An acceleration malfunction detection circuit performs self checks of incremental velocity pulses and critical voltage. Incremental velocity pulses from each of the three axes are checked for presence. If pulses are absent longer than 0.35 seconds, it indicates that a flip flop did not reset between set pulses.

The critical voltage (+12V DC) is checked for presence. If a malfunction is detected, an ACC light on the control panel is automatically illuminated. If momentary malfunctions occur, the accelerometer malfunction circuit can be restored to normal operation by pressing the RESET button.

NOTE

Malfunction of the accelerometer circuits does not affect attitude measurements.

AUXILIARY COMPUTER POWER UNIT

The Auxiliary Computer Power Unit (ACPU) is used in conjunction with the IGS power supply to maintain the correct DC voltages at the computer. The computer cannot function properly on low voltage, either as a transient or a depression. Abnormal voltages can cause permanent changes in the computer memory. Three types of circuits are provided in the ACPU to prevent a low voltage condition at the computer. The first circuit is a transient sense and auxiliary power control circuit. The second circuit is a low voltage sense and power control circuit and the third is auxiliary power. The ACPU is turned on and off with the computer power switch.

Transient Sense Circuit

The transient sense circuit is designed to sense and correct transient low voltage conditions. A series type transistor voltage regulator holds auxiliary power off the line as long as IGS power supply, computer voltage regulator, voltage is normal. If regulator voltage momentarily drops below 17.7 volts, the transient sense circuit detects the drop and turns on the series regulator. The regulator then places auxiliary power on the line and maintains voltage at the desired level.

Low Voltage Sense Circuit

A low voltage sense circuit prevents the computer from operating on low voltage. When the computer is turned on, the low voltage sense circuit insures that spacecraft bus voltage is above 21 volts before allowing power to be applied to the computer. If the computer is already on when a low voltage condition occurs, the transient sense circuit will maintain normal voltage for 100 milliseconds. If spacecraft bus voltage is not back to normal after 100 milliseconds, the low voltage sense circuit initiates a controlled shutdown of the computer. Computer power is controlled through contacts of a relay in the low voltage sense circuit. When the low voltage sense circuit detects a voltage depression, it de-energizes the relay. Contacts of the relay turn off the computer in a manner identical with the computer power switch. When the low voltage sense circuit turns off the computer, it also breaks power to all ACPU circuits except low voltage sense. If power were not broken to the transient sense circuit, it would attempt to maintain normal voltage at the computer. In attempting to maintain normal voltage, the auxiliary power capability would be exceeded.

Auxiliary Power

Auxiliary power consists of a battery and a trickle charger. A 0.5 ampere-hour nickle cadmium battery is used to supply computer power during spacecraft bus low voltage transients. The battery will supply up to 9.8 amperes for periods of 100 milliseconds or less. A trickle charger is provided to maintain a full charge on the battery. The charger consists of a transistor oscillator, transformer, and rectifier. The oscillator changes spacecraft bus voltage to AC. The AC voltage is then stepped up with a transformer and changed back to DC by a full wave diode rectifier. Rectifier output is then applied, through a current limiting resistor, to the battery. The resistor limits charging current to 25 milliamperes. Provision is included to charge the battery from an external source if desired.

DIGITAL COMPUTER

SYSTEM DESCRIPTION

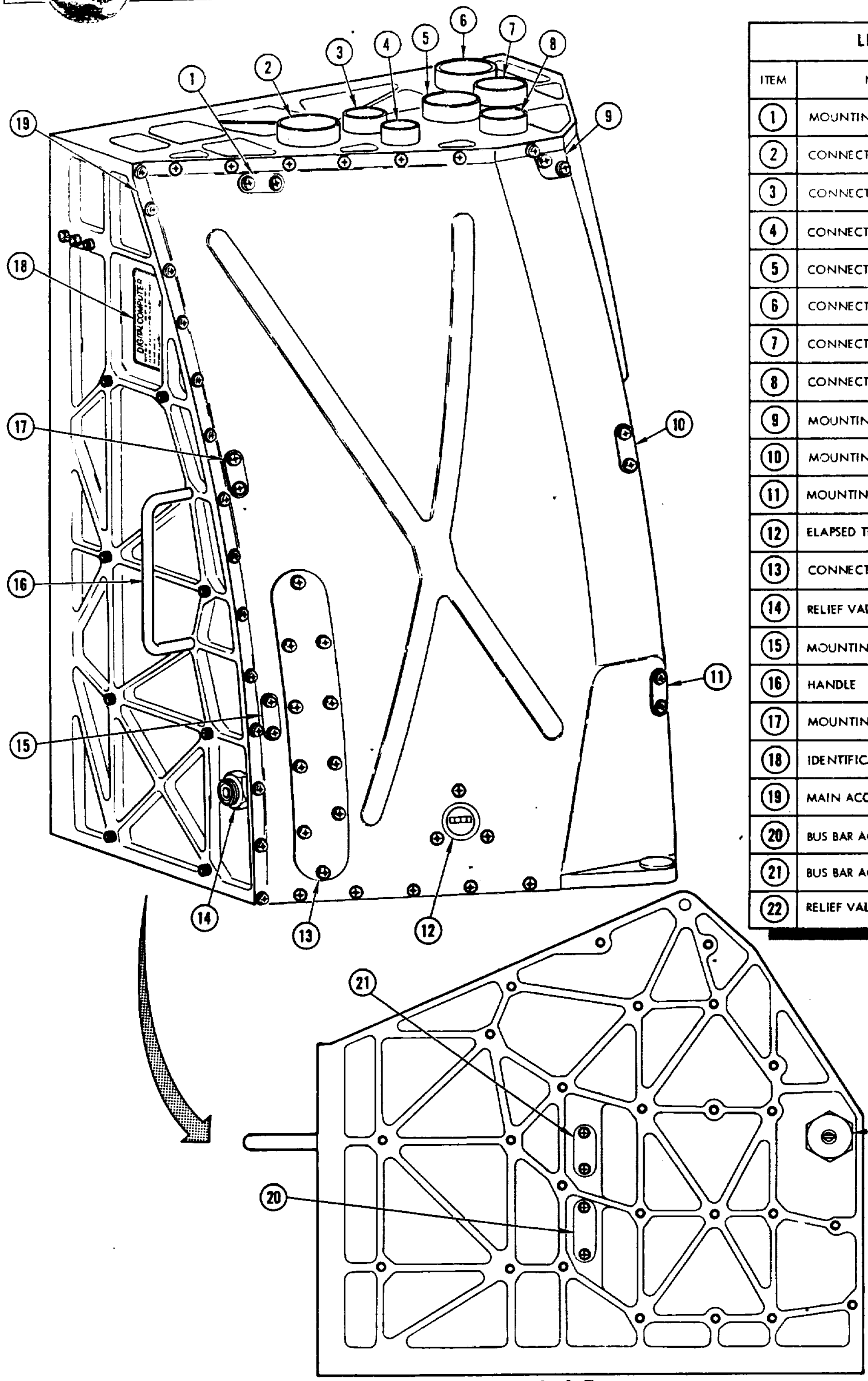
General

The Digital Computer, hereinafter referred to as the computer, is a binary, fixed-point, stored-program, general-purpose computer, used to guide the spacecraft. The computer is 18.90 inches high, 14.50 inches wide, and 12.75 inches deep. It weighs 58.98 pounds. External views of the computer are shown on Figure 8-20. The major external characteristics are summarized in the accompanying legend.

Using inputs from other spacecraft systems along with a stored program, the computer performs the computations necessary to develop the guidance and control outputs required by the spacecraft during the Pre-Launch and Re-Entry phases of the mission. In addition, the computer provides back-up guidance for the launch vehicle during Ascent.

Inputs and Outputs

The computer is interfaced with the Inertial Platform, Platform Electronics, Inertial Guidance System (IGS) Power Supply, Auxiliary Computer Power Unit (ACPU), Manual Data Insertion Unit (MDIU), Time Reference System (TRS), Digital Command System (DCS), Attitude Display, Attitude Control and Maneuver Electronics (ACME), Titan Autopilot, Pilots' Control and Display Panel (PCDP), Incremental Velocity Indicator (IVI), Instrumentation System (IS), and Aerospace Ground Equipment (AGE). In connection with these interfaces, the computer inputs and outputs include the following:



LEGEND	
ITEM	NOMENCLATURE
1	MOUNTING ACCESS COVER
2	CONNECTOR J4
3	CONNECTOR J5
4	CONNECTOR J7
5	CONNECTOR J3
6	CONNECTOR J2
7	CONNECTOR J1
8	CONNECTOR J6
9	MOUNTING ACCESS COVER
10	MOUNTING ACCESS COVER
11	MOUNTING ACCESS COVER
12	ELAPSED TIME INDICATOR
13	CONNECTOR ACCESS COVER
14	RELIEF VALVE
15	MOUNTING ACCESS COVER
16	HANDLE
17	MOUNTING ACCESS COVER
18	IDENTIFICATION PLATE
19	MAIN ACCESS COVER
20	BUS BAR ACCESS COVER
21	BUS BAR ACCESS COVER
22	RELIEF VALVE

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Figure 8-20 Digital Computer

Inputs

40 discrete

3 incremental velocity

3 gimbal angle

2 high-speed data (500 kc)

1 low-speed data (3.57 kc)

1 low-speed data (182 cps)

1 input and readback (99 words)

6 DC power (5 regulated, 1 unregulated)

1 AC power (regulated)

Outputs

30 discrete

3 steering command

3 incremental velocity

1 decimal display (7 digits)

1 telemetry (21 digital data words)

1 low-speed data (3.57 kc)

1 low-speed data (182 cps)

3 DC power (regulated)

1 AC power (regulated, filtered)

Operational Characteristics

The major operational characteristics of the computer are as follows:

Type

Binary, fixed-point, stored-program, general-purpose

Memory

Random-access, nondestructive-readout

Flexible division between instruction and data storage

4096 addresses, 39 bits per address

13 bits per instruction word

26 bits per data word

Arithmetic Times

Instruction cycle - 140 usec

Divide requires 6 cycles

Multiply requires 3 cycles

All other instructions require 1 cycle each

Other instructions can be programmed concurrently with multiply and divide

Clock Rates

Arithmetic bit rate - 500 kc

Memory cycle rate - 250 kc

Controls and Indicators

The computer itself contains no controls or indicators, with the exception of the elapsed time indicator. However, the computer can be controlled by means of four switches located on the Pilots' Control and Display Panel: the two-position Computer On-Off switch, the seven-position Computer Mode switch, the push-button Start Computation switch, and the push-button Malfunction Reset switch.

SYSTEM OPERATION

Power

The computer receives the AC and DC power required for its operation from the Inertial Guidance System (IGS) Power Supply. The regulated DC power supplied to the computer is buffered in the IGS Power Supply in a manner that eliminates any loss in regulation due to transients that occur in the spacecraft prime power source. Actual power interruptions and depressions are buffered by the IGS Power Supply and the Auxiliary Computer Power Unit. The power inputs received from the IGS Power Supply are as follows:

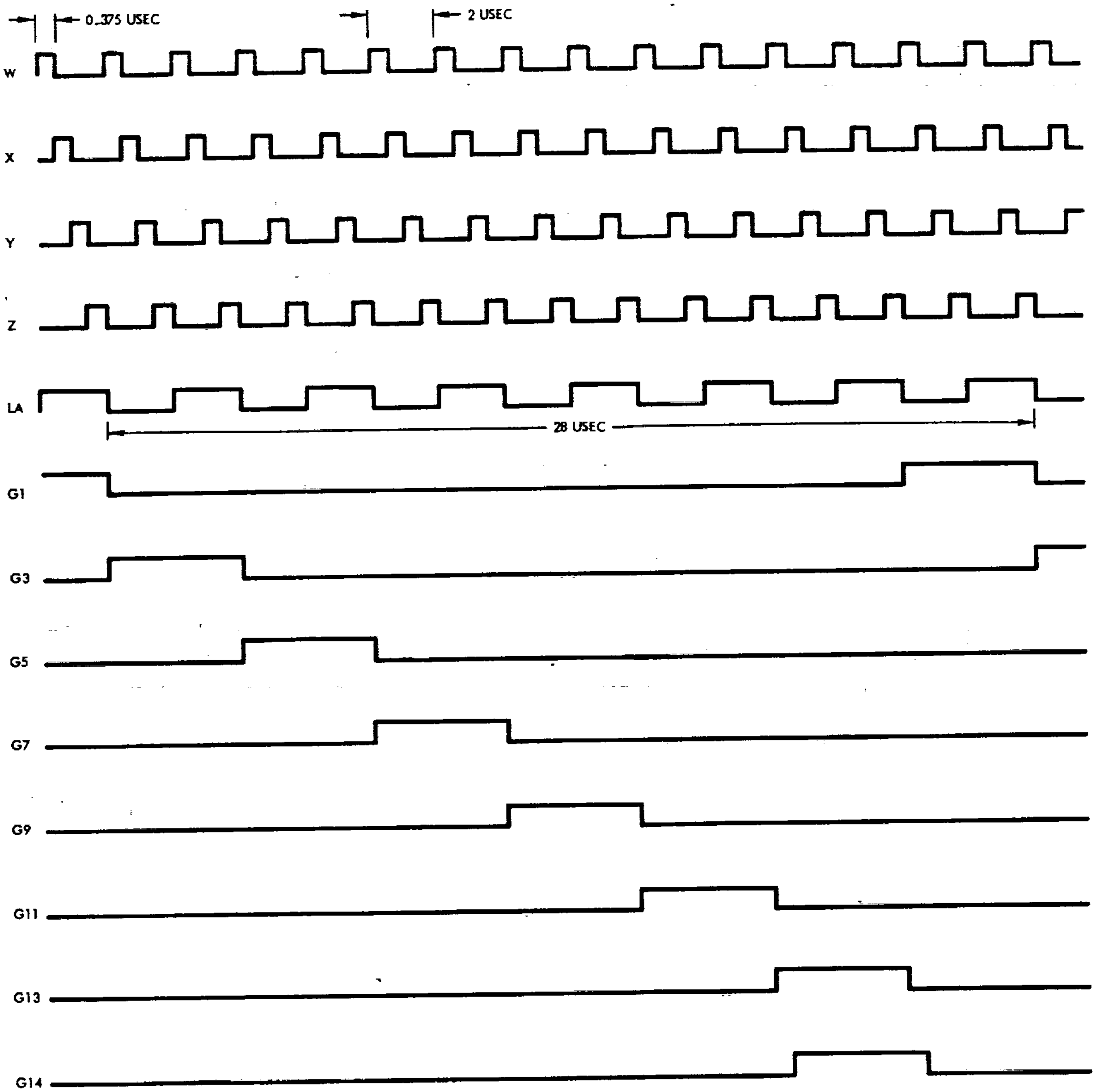
(a) 26 VAC and return
(b) +28 VDC filtered and return
(c) +27.2 VDC and return
(d) -27.2 VDC and return
(e) +20 VDC and return
(f) +9.3 VDC and return

The application of all power is controlled by the Computer On-Off switch on the Pilots' Control and Display Panel. When the switch is turned on, the computer elapsed time indicator starts operating and a power control signal is supplied to the IGS Power Supply by the computer. This signal causes power to be transferred to the computer. When the switch is turned off, the computer elapsed time indicator stops operating and the power control signal is terminated to remove power from the computer.

Within the computer, the 26 VAC power is used by magnetic modulators to convert DC analog signals to AC analog signals. This power is also used by a harmonic filter to develop a 16 VAC, 400 cps filtered gimbal angle excitation signal. The +28 VDC power is used by computer power sequencing circuits. The +27.2 VDC, -27.2 VDC, +20 VDC, and +9.3 VDC power is used by power regulators to develop +25 VDC, -25 VDC, and +8 VDC regulated power. This regulated power is used by logic circuits throughout the computer.

Basic Timing

The basic computer timing is derived from an 8 mc oscillator. The 8 mc signal is counted down to generate four clock pulses (called W, X, Y, and Z) (Figure 8-21). These clock pulses are the basic timing pulses from which all other timing is generated. The width of each clock pulse is 0.375 usec and the pulse repetition frequency is 500 kc. The bit time is 2 usec, and a new bit time is considered as starting each time the W clock pulse starts. Eight gate signals (G1, G3, G5, G7, G9, G11, G13, and G14) are generated, each lasting two bit times. The first and second bit times of a particular gate are discriminated by use of a control signal (called LA) which is on for odd bit times and off for even bit times. Fourteen bit times make up one phase time, resulting in a phase time length of 28 usec (Figure 8-22). Five phases (PA through PE) are required to complete a computer instruction cycle, resulting in an instruction cycle length of 140 usec. Special phase timing, consisting of four phases (PH1 through PH4) (Figure 8-23), is generated for use by the input processor and the output processor. This timing is independent of computer phase timing but is synchronized with computer bit timing.



BIT TIMING TABLE

BIT TIME (BT)	LA	GATE	BIT TIME (BT)	LA	GATE	BIT TIME (BT)	LA	GATE
1	"1"	G1	6	"0"	G7	11	"1"	G11
2	"0"	G3	7	"1"	G7	12	"0"	G13
3	"1"	G3	8	"0"	G9	13	"1"	G13
4	"0"	G5	9	"1"	G9	14	"0"	G1
5	"1"	G5	10	"0"	G11			

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Figure 8-21 Computer Clock and Bit Timing

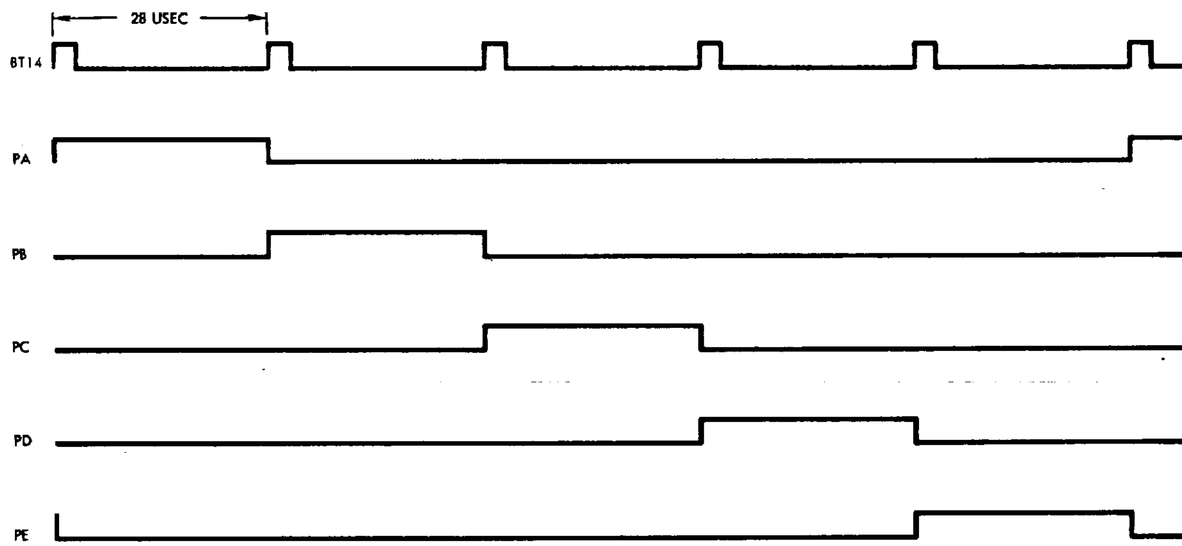

Figure 8-22 Computer Phase Timing

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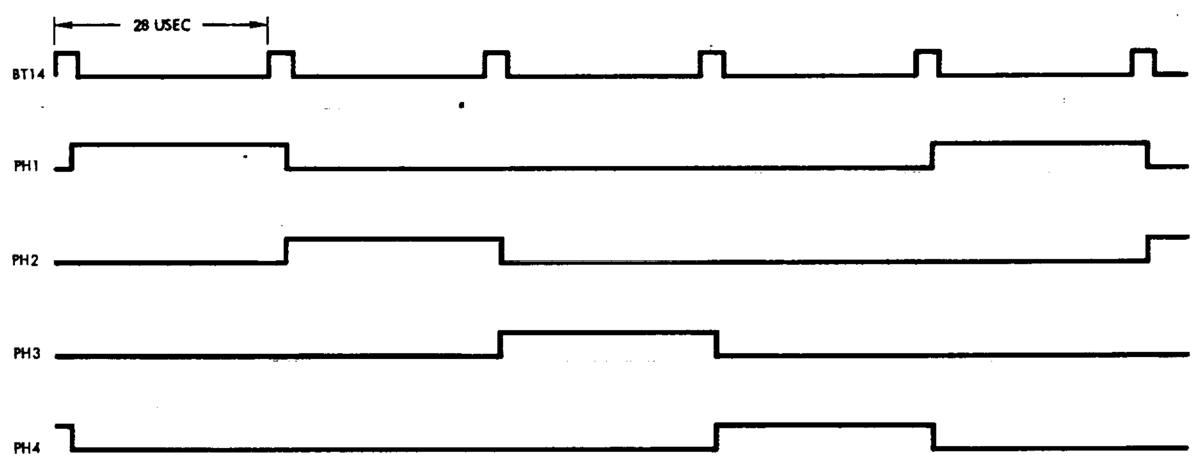

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Figure 8-23 Processor Phase Timing

Memory

The computer memory is a random-access, coincident-current, ferrite array with nondestructive readout. The basic storage element is a two-hole ferrite core. The nondestructive read property makes it possible to read or write serially or in series-parallel, thereby allowing operation with a serial arithmetic unit without a separate buffer register. The memory array can store 4096 words, or 159,744 bits. All memory words of 39 bits are divided into three syllables of 13 bits each. Data words (25 bits and a sign) are normally stored in the first two syllables, and instruction words (13 bits) are intermixed in all three syllables. Once the spacecraft has been removed from the hangar area, it is not possible to modify the third syllable of any memory word. Limited modification of stored data in syllables 0 and 1 can be accomplished at the launch site through interface with the Manual Data Insertion Unit or the Digital Command System.

As shown on Figure 8-24, the memory is a 64 x 64 x 39 bit array of nondestructive readout elements. Physically, it consists of a stack of 39 planes (stacked in the Z dimension), with each plane consisting of a 64 x 64 array of cores. The memory is logically subdivided into smaller parts to increase the program storage efficiency. The Z dimension is divided into three syllables (SYL 0 through SYL 2), with each syllable consisting of 13 bits. The X-Y plane is divided into 16 sectors (SEC 00 through SEC 07, and SEC 10 through SEC 17), with sector 17 being defined as the residual sector.

A memory word is defined as the 39 bits along the Z dimension and is located at one of the 4096 possible X-Y grid positions. An instruction word or command requires 13 bits, and is coded in either syllable 0, 1, or 2 of a memory word.

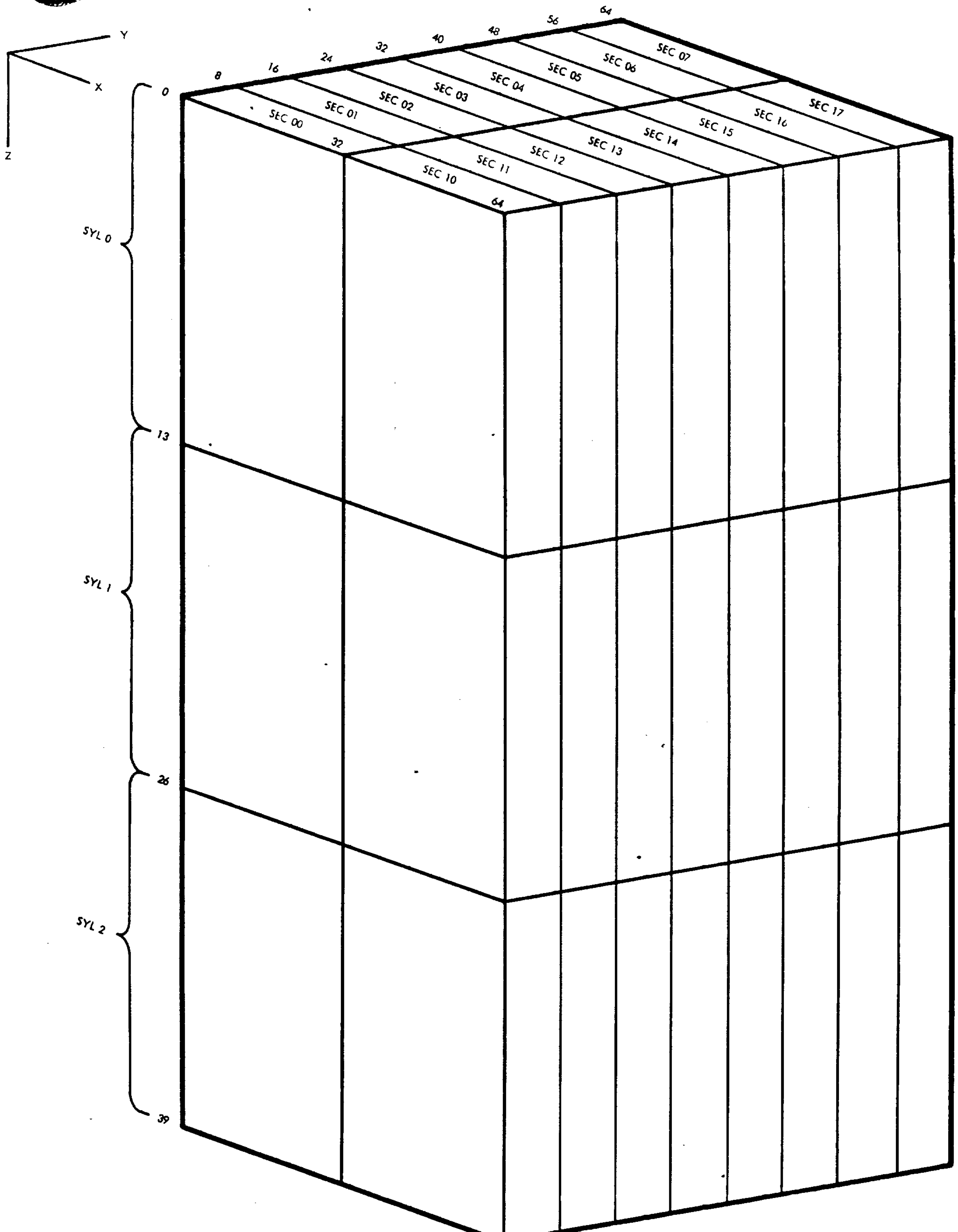


Figure 8-24 Computer Memory Functional Organization

A data word requires 26 bits, and is always coded in syllables 0 and 1 of a memory word. Information stored in syllable 2 can be read as a short data word by using a special mode of operation primarily used to check the contents of the memory.

NOTE

The operation codes mentioned in the subsequent paragraphs are described in the Instruction and Data Words paragraph.

Instruction List

The instructions which can be executed by the computer are as follows:

Operation Code	Instruction
0000	HOP. The contents of the memory location specified by the operand address are used to change the next instruction address. Four bits identify the next sector, nine bits are transferred to the instruction address counter, two bits are used to condition the syllable register, and one bit is used to select one of the two data word modes.
0001	DIV (divide). The contents of the memory location specified by the operand address are divided by the contents of the accumulator. The 24-bit quotient is available in the quotient delay line during the fifth word time following the DIV.

Operation Code (cont)	Instruction (cont)
0010	PRO (process input or output). The input or output specified by the operand address is read into, or loaded from, the accumulator. An output command clears the accumulator to zero if address bit A9 is a "1." The accumulator contents are retained if A9 is a "0." (Refer to Table 8-1 for a list of the PRO instructions.)
0011	RSU (reverse subtract). The contents of the accumulator are subtracted from the contents of the specified memory location. The result is retained in the accumulator.
0100	ADD. The contents of the memory location specified by the operand address are added to the contents of the accumulator. The result is retained in the accumulator.
0101	SUB (subtract). The contents of the memory location specified by the operand address are subtracted from the contents of the accumulator. The result is retained in the accumulator.
0110	CLA (clear and add). The contents of the memory location specified by the operand address are transferred to the accumulator.

Operand Address		Signal
X (Bits A1-A3)	Y (Bits A4-A6)	
0	0	Digital command system shift pulse gate
0	1	Instrumentation system control gate
0	2	Time reference system data and timing pulses
0	3	Digit magnitude weight 1
0	4	Reset data ready, enter, and readout
0	5	Digit select weight 1
0	6	Memory strobe
1	0	Computer ready
1	1	Drive counters to zero
1	2	Enter
1	3	Digit magnitude weight 2
1	4	Display device drive
1	5	Digit select weight 2
1	6	Autopilot scale factor
2	0	Pitch resolution
2	1	Select X counter
2	2	Aerospace ground equipment data link
2	3	Digit magnitude weight 4
2	5	Digit select weight 4
2	6	Reset start computation

Table 8-1. PRO Instruction Programming (1 of 3)

Operand Address		Signal
X (Bits A1-A3	Y (Bits A4-A6)	
3	0	Yaw resolution
3	1	Select Y counter
3	2	Aerospace ground equipment data clock
3	3	Digit magnitude weight 8
3	4	Read manual data insertion unit insert data
3	6	Reset radar ready
4	0	Roll resolution
4	1	Elapsed time control and time reference system control reset
4	3	Computer malfunction
4	4	Spare
4	6	Second stage engine cutoff
5	0	Computer running
5	1	Time to start re-entry calculations control
5	2	Time to reset control
5	3	Write output processor
5	4	Read delta velocity
5	5	Input processor time
5	6	Time to retrofire control
6	3	Read pitch gimbal
6	4	Read roll gimbal

Table 8-1. PRO Instruction Programming (2 of 3)

Operand Address		Signal
X (Bits A1-A3)	Y (Bits A4-A6)	
6	5	Read yaw gimbal
7	0	Pitch error command
7	1	Yaw error command
7	2	Roll error command

Table 8-1. PRO Instruction Programming (3 of 3)

Operation Code (cont)	Instruction (cont)
0111	AND. The contents of the memory location specified by the operand address are logically ANDed, bit-by-bit, with the contents of the accumulator. The result is retained in the accumulator.
1000	MPY (multiply). The contents of the memory location specified by the operand address are multiplied by the contents of the accumulator. The 24 high-order bits of the multiplier and multiplicand are multiplied together to form a 26-bit product which is available in the product delay line during the second word time following the MPY.
1001	TRA (transfer). The operand address bits (A1 through A9) are transferred to the instruction address counter to form a new instruction address. The syllable and sector remain unchanged.
1010	SHF (shift). The contents of the accumulator are shifted left or right, one or two places, as specified by the operand address, according to the following table:

Operation Code (cont)	Instruction (cont)		
	Command	Operand Address X (Bits A1-A3)	Operand Address Y (Bits A4-A6)
1010 (cont)	Shift left one place	*	3
	Shift left two places	*	4
	Shift right one place	1	2
	Shift right two places	0	2

* Insignificant

If an improper address code is given, the accumulator is cleared to zero. While shifting left, "0's" are shifted into the low-order positions; while shifting right, the sign bit condition is shifted into the high-order positions.

1011 TMI (transfer on minus accumulator sign). If the sign is positive ("0"), the next instruction in sequence is chosen (no branch). If the sign is negative ("1"), the nine bits of operand address become the next instruction address (perform branch). The syllable and sector remain unchanged.

1100 STO (store). The contents of the accumulator are stored in the memory location specified by the operand address. The contents of the accumulator are also retained for later use.

1101 SPQ (store product or quotient). The product is available on the second word time following an MPY. The quotient is available on the fifth word time following a DIV. The product or quotient is stored in the memory location specified by the operand address.

1110 CLD (clear and add discrete). The state of the discrete input selected by the operand address is read into all accumulator bit positions. (Refer to Table 8-2 for a list of the CLD instructions.)

1111 TNZ (transfer on non-zero). If the contents of the accumulator are zero, the next instruction in sequence is chosen (no branch); if the contents are non-zero, the nine bits of operand address become the next instruction address (perform branch). The syllable and sector remain unchanged.

NOTE

The instructions mentioned in the subsequent paragraphs (e.g., HOP, TRA, TMI, and TNZ) are described more completely in the Instruction Information Flow paragraph.

Operand Address		Signal
X (Bits A1-A3)	Y (Bits A4-A6)	
0	0	Radar ready
0	1	Computer mode 2
0	2	Spare
0	3	Processor timing phase 1
0	4	Spare
1	0	Data ready
1	1	Computer mode 1
1	2	Start computation
1	3	X zero indication
1	4	Spare
2	0	Enter
2	1	Instrumentation system sync
2	2	Velocity error count not zero
2	3	Aerospace ground equipment request
2	4	Spare
3	0	Readout
3	1	Computer mode 3
3	2	Spare
3	3	Spare
3	4	Spare
4	0	Clear

Table 8-2. CLD Instruction Programming (1 of 2)

Operand Address		Signal
X (Bits A1-A3)	Y (Bits A4-A6)	
4	1	Spare
4	2	Simulation mode command
4	3	Spare
4	4	Spare
5	0	Time to start re-entry calculations
5	1	Spare
5	2	Y zero indication
5	3	Spare
5	4	Spare
6	0	Digital command system ready
6	1	Fade-in discrete
6	2	Z zero indication
6	3	Umbilical disconnect
6	4	Spare
7	0	Instrumentation system request
7	1	Abort transfer
7	2	Aerospace ground equipment input data
7	3	Spare
7	4	Spare

Table 8-2. CLD Instruction Programming (2 of 2)

Instruction Sequencing

The instruction address is derived from an instruction counter and its associated address register. To address an instruction, the syllable, sector, and word position within the sector (one of 256 positions) must be defined. The syllable and sector are defined by the contents of the syllable register (two-bit code, three combinations) and sector register (four-bit code, 16 combinations). These registers can be changed only by a HOP instruction. The word position within the sector is defined by the instruction address counter. The instruction address count is stored serially in a delay line; and normally each time it is used to address a new instruction, a one is added to it so that the instruction locations within a sector can be sequentially scanned. The number stored in the counter can be changed by either a TRA, TMI, or TNZ instruction, with the operand address specifying the new number. A HOP instruction can also change the count, with the new instruction location coming from a data word.

Instruction and Data Words

The instruction word consists of 13 bits and can be coded in any syllable of any memory word. The word is coded as follows:

Bit Position	1	2	3	4	5	6	7	8	9	10	11	12	13
Bit Code	A1	A2	A3	A4	A5	A6	A7	A8	A9	OP1	OP2	OP3	OP4

The four operation bits (OP1 through OP4) define one of 16 instructions, the eight operand address bits (A1 through A8) define a memory word within the sector being presently used, and the residual bit (A9) determines whether or not to read the data residual. If the A9 bit is a "1," the data word addressed is always located in the last sector (sector 17). If the A9 bit is a "0," the data word addressed is read from the sector defined by the contents of the sector register. This feature allows data locations to be available to instructions stored anywhere in the memory.

The data word consists of 25 magnitude bits and a sign bit. Numbers are represented in two's-complement form, with the low-order bits occurring at the beginning of the word and the sign bit occurring after the highest-order bit. The binary point is placed between bit positions 25 and 26. The bit magnitude number also denotes the binary weight of the position. For example, M16 represents 2^{-16}. For the HOP instruction, the next instruction address is coded in a data word that is read from the memory location specified by the operand address of the HOP word. The codings of a numerical data word and a HOP word are as follows:

Bit Position	1	2	3	4	5	6	7	8	9	10	11	12	13
Data Word	M25	M24	M23	M22	M21	M20	M19	M18	M17	M16	M15	M14	M13
HOP Word	A1	A2	A3	A4	A5	A6	A7	A8	A9	S1	S2	S3	S4

Bit Position	14	15	16	17	18	19	20	21	22	23	24	25	26
Data Word	M12	M11	M10	M9	M8	M7	M6	M5	M4	M3	M2	M1	S
HOP Word	-	SYA	SYB	-	S5	-	-	-	-	-	-	-	-

For the HOP word, eight address bits (A1 through A8) select the next instruction (one of 256) within the new sector, the residual bit (A9) determines whether or not the next instruction is located in the residual sector, the sector bits

(S1 through S4) select the new sector, and the syllable bits (SYA and SYB) select the new syllable according to the following table:

Syllable	SYB	SYA
0	"0"	"0"
1	"0"	"1"
2	"1"	"0"

The special syllable bit (S5) determines the mode in which data words are to be read. If the S5 bit is a "0," normal operation of reading data words from syllables 0 and 1 is followed; however, if the S5 bit is a "1," data words are read from syllable 2 only. These data words contain information from syllable 2 in bit positions 1 through 13, but contain all "0's" in bit positions 14 through 26. This special mode is followed until a new HOP command places the computer back in the normal mode of reading data words. (While in the special mode, any HOP word addressed always has "0's" coded in the SYA, SYB, and S5 positions due to the short data word that is read; therefore, any HOP word coded while in this mode terminates the mode and operation is resumed in syllable 9.) The computer itself does not have the capability to store information in syllable 2; therefore, STO and SPQ commands are not executed while in the special mode. The mode is used only to allow the computer arithmetic circuits to check the entire memory contents to verify the fact that the proper information is in storage.

In a HOP word, the residual bit (A9) overrides the sector bits (S1 through S4). If the A9 bit is a "1," the next instruction is read from the residual sector.

If, however, the A9 bit is a "0," the S1 through S4 bits determine the sector from which the next instruction is read.

For convenience, the data and instruction words can be coded in an octal form that is easily converted to the machine binary representation. The order in which the bits are written is reversed to conform to the normal method of placing lower-significance bits to the right. (The computer words are organized with lower-significance bits to the left so that, while performing arithmetic, the low-order bits are accessed first.) The coding structure is as follows:

Instruction Word

OP4	OP3 OP2 OP1	A9 A8 A7	A6 A5 A4	A3 A2 A1
			*Y Address	*X Address

*Addresses for CLD and PRO instructions

Data Word

S M1 M2	M3 M4 M5	M6..........M20	M21 M22 M23	M24 M25

where each group of three bits is expressed as an octal character (from 0 to 7). An instruction word is thus expressed as a five-character octal number. The operation code can take on values from 00 to 17, and the operand address can take on values from 000 to 777. Any operand address larger than 377 addresses the residual sector (sector 17) because the highest-order address bit (A9) is also the residual identification bit. A data word is expressed as a nine-character octal number, taking on values from 000000000 to 777777776. The low-order character can take on only the values of 0, 2, 4, and 6.

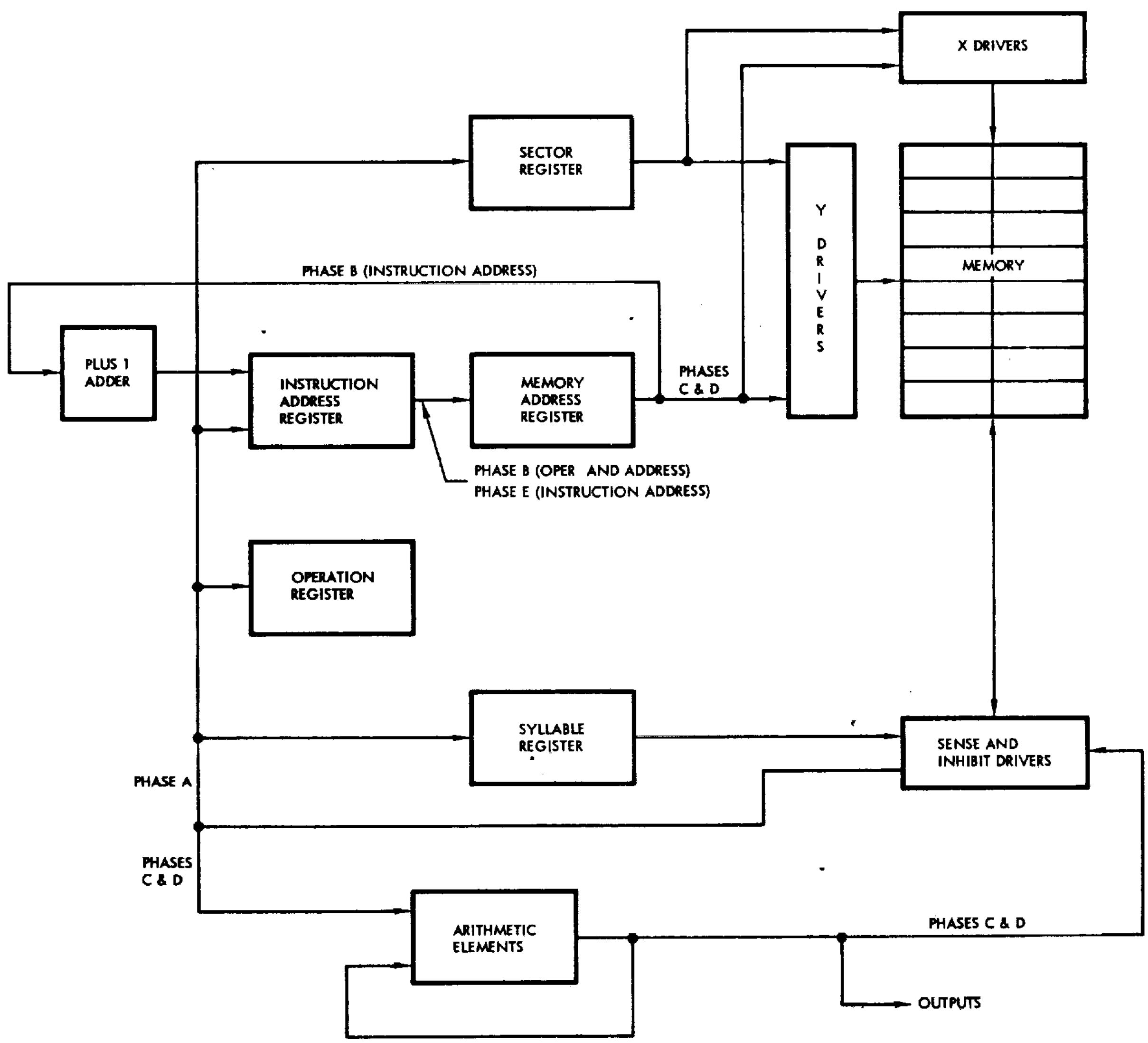
Arithmetic Elements

The computer has two arithmetic elements: an add-subtract element (accumulator), and a multiply-divide element. Each element operates independently of the other; however, both are serviced by the same program control circuits. Computer operation times can be conveniently defined as a number of cycles, where a cycle time represents the time required to perform an addition (140 usec). All operations except MPY and DIV require one cycle; MPY requires three cycles, and DIV requires six cycles. Each cycle, the program control is capable of servicing one of the arithmetic elements with an instruction. An MPY or a DIV instruction essentially starts an operation in the multiply-divide element, and the program control must obtain the answer at the proper time since the multiply-divide element has no means of completing an operation by itself. When an MPY is commanded, the product is obtainable from the multiply-divide element two cycle times later by an SPQ instruction. When a DIV is commanded, the quotient is obtainable five cycle times later by an SPQ instruction.

It is possible to have one other instruction run concurrently between the MPY and the SPQ during multiply, and four other instructions run concurrently between the DIV and the SPQ during divide. However, an MPY or a DIV is always followed with an SPQ before a new MPY or DIV is given.

Basic Information Flow

Refer to Figure 8-25 for the following description of information flow during the five computer phase times. The description is limited to those operations requiring only one cycle time, and thus does not pertain to MPY and DIV.



FM2-8-25

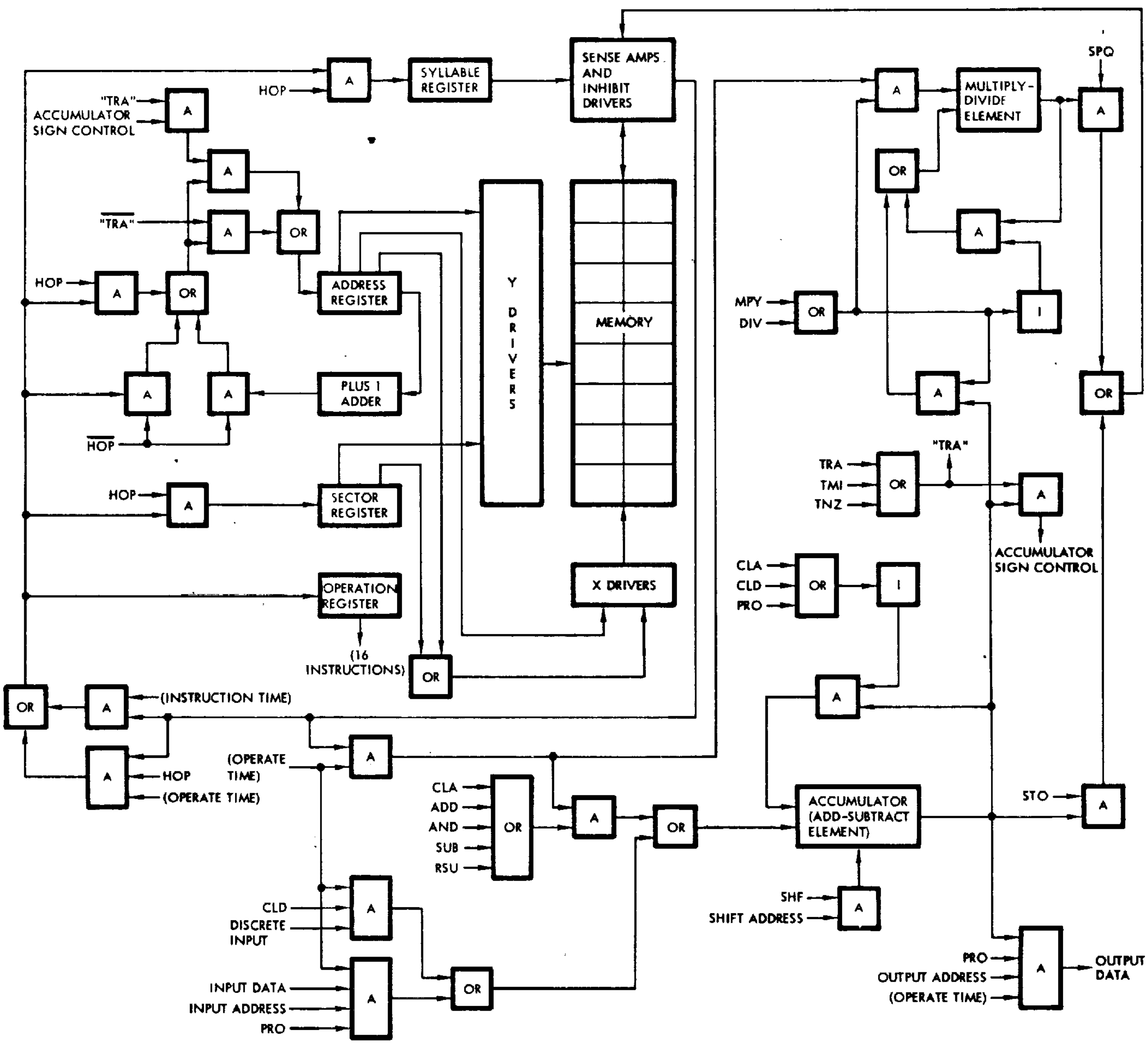
Figure 8-25 Basic Information Flow

During phase A, the 13-bit instruction word is read from memory and stored in the instruction address register. The address of the instruction is defined by the contents of the memory address register, the sector register, and the syllable register. The four operation code bits (OP1 through OP4) are stored in the operation register. During phase B, the operand address bits (A1 through A8) are serially transferred from the instruction address register to the memory address register. Simultaneously, the instruction address stored in the memory address register is incremented by plus one and stored in the instruction address register. The operation specified by the operation code bits is performed during phases C and D. During phase E, the next instruction address, stored in the instruction address register, is transferred to the memory address register.

Four of the one-cycle operations do not strictly adhere to the above information flow. These operations are HOP, TRA, TMI, and TNZ. For the HOP instruction, data read from memory during phases C and D is transferred directly to the instruction address register, the sector register, and the syllable register. For the TRA, TMI, and TNZ operations, the transfer of the next instruction address from the instruction address register during phase E is inhibited to allow the operand address to become the next instruction address.

Instruction Information Flow

Flow Diagram: The instruction information flow diagram (Figure 8-26) should be used along with the following descriptions.



NOTE

A= AND; I = INVERTER

Figure 8-26 Instruction Information Flow

FM2-8-26

CLA Operation

During phases C and D, the data that was contained in the accumulator during phases A and B is destroyed. Simultaneously, new data from the selected memory location is transferred through the sense amplifiers and into the accumulator. During phases E and A, the new data is recirculated so as to be available in the accumulator during phases A and B.

ADD Operation

During phases C and D, new data from the selected memory location is transferred through the sense amplifiers and into the accumulator. Here, the new data is added to the data that was contained in the accumulator during phases A and B. During phases E and A, the sum data is recirculated so as to be available in the accumulator during phases A and B.

SUB Operation

During phases C and D, new data from the selected memory location is transferred through the sense amplifiers and into the accumulator. Here, the new data is subtracted from the data that was contained in the accumulator during phases A and B. During phases E and A, the difference data is recirculated so as to be available in the accumulator during phases A and B.

RSU Operation

During phases C and D, new data from the selected memory location is transferred through the sense amplifiers and into the accumulator. Here, the data that was contained in the accumulator during phases A and B is subtracted from the new data. During phases E and A, the difference data is recirculated so as to be available in the accumulator during phases A and B.

AND Operation

During phases C and D, new data from the selected memory location is transferred through the sense amplifiers and into the accumulator. Here, the new data is ANDed with the data that was contained in the accumulator during phases A and B. During phases E and A, the ANDed data is recirculated so as to be available in the accumulator during phases A and B.

SHF Operation

During phases C and D, the data that was contained in the accumulator during phases A and B is shifted left or right, one or two places, as specified by the operand address. During phases E and A, the shifted data is recirculated so as to be available in the accumulator during phases A and B.

STO Operation

During phases C and D, the data that was contained in the accumulator during phases A and B is transferred through the inhibit drivers and stored in the memory location selected by the operand address. During phases E and A, the same data is recirculated so as to be available in the accumulator during phases A and B.

HOP Operation

During phases C and D, new data from the selected memory location is transferred through the sense amplifiers and into the address, sector, and syllable registers. Here, the new data is used to select the address, sector, and syllable of the memory location from which the next instruction will be read.

TRA Operation

During phases A and B, the instruction from the selected memory location is transferred through the sense amplifiers and into the address register. Here, the instruction is used to select the address of the memory location from which the next instruction will be read. The sector and syllable remain unchanged.

TMI Operation

During phases A and B, the instruction from the selected memory location is transferred through the sense amplifiers and into the address register. Here, if the accumulator sign is negative, the instruction is used to select the address of the memory location from which the next instruction will be read. However, if the accumulator sign is positive, the next instruction address in sequence is selected in the normal manner. The sector and syllable remain unchanged.

TNZ Operation

During phases A and B, the instruction from the selected memory location is transferred through the sense amplifiers and into the address register. Here, if the contents of the accumulator are not zero, the instruction is used to select the address of the memory location from which the next instruction will be read. However, if the contents of the accumulator are zero, the next instruction address in sequence is selected in the normal manner. The sector and syllable remain unchanged.

CLD Operation

During phases C and D, the data that was contained in the accumulator during phases A and B is destroyed. Simultaneously, the state of the discrete input selected by the operand address is transferred into all accumulator bit positions. During phases E and A, the new data is recirculated so as to be available in the accumulator during phases A and B.

PRO Operation (Inputs; When A9="1")

During phases C and D, the data that was contained in the accumulator during phases A and B is destroyed. Simultaneously, the data on the input channel selected by the operand address is transferred into the accumulator. During phases E and A, the new data is recirculated so as to be available in the accumulator during phases A and B.

PRO Operation (Inputs; When A9="0")

During phases C and D, the data on the input channel selected by the operand is transferred into the accumulator. Here, the new data is ORed with the data that was contained in the accumulator during phases A and B. During phases E and A, the ORed data is recirculated so as to be available in the accumulator during phases A and B.

PRO Operation (Outputs)

During phases C and D, the data that was contained in the accumulator during phases A and B is transferred to the output channel selected by the operand address. If the A9 of the operand address is a "1," the data that was contained in the accumulator during phases A and B is then destroyed. However, if the A9 bit is a "0," the data is recirculated so as to be available in the accumulator during phases A and B.

MPY Operation

During phases A and B of the first instruction cycle, the data that is contained in the accumulator is transferred into the multiply-divide element as the multiplier. During phases C and D of the same cycle, new data from the selected memory location is transferred through the sense amplifiers and into the multiply-divide element as the multiplicand. During the remainder of the first instruction cycle and the next two instruction cycles, the multiplicand is multiplied by the multiplier. The product is available in the mulitply-divide element during phases C and D of the third instruction cycle.

DIV Operation

During phases A and B of the first instruction cycle, the data that is contained in the accumulator is transferred into the multiply-divide element as the divisor. During phases C and D of the same cycle, new data from the selected memory address is transferred through the sense amplifiers and into the multiply-divide element as the dividend. During the remainder of the first instruction cycle and the next five instruction cycles, the dividend is divided by the divisor. The quotient is available in the multiply-divide element during phases C and D of the sixth instruction cycle.

SPQ Operation

During phases C and D, the product or quotient that is contained in the multiply-divide element is transferred through the inhibit drivers and stored in the memory location selected by the operand address.

NOTE

In the subsequent program and interface descriptions, the signals that are programmed by CLD and PRO instructions are sometimes referred to as DI (discrete input) or DO (discrete output) signals. The two digits following the DI or DO are the Y and X addresses, respectively, of the instruction.

Operational Program

The operational program consists of six basic routines: Executor, Pre-Launch, Ascent, Catch-Up, Rendezvous and Re-Entry (Catch-Up & Rendezvous not applicable for S/C 3, 4 & 7). Each routine is made up of several subroutines. Some of the subroutines are common to all routines while some are unique to a particular routine. Each subroutine consists of a series of program instructions which, when executed, cause specific computer circuits to operate. The initiation of a particular routine is controlled by the Computer Mode switch on the Pilots' Control and Display Panel. Once a routine is initiated, the subroutines within the routine are executed automatically.

Executor Routine

The Executor routine selects and handles the functions common to all other routines. The program flow for this routine is shown on Figure 8-27. The individual blocks shown on the figure are explained as follows:

(a) Block 1. When the computer is turned on, the first memory location addressed is address 000, sector 00, syllable 0. This memory location is the first memory address utilized by the Executor routine.

(b) Block 2. The operational program utilizes special predetermined memory locations which are designed as logical choice (LC) addresses. At certain times, the sign bits at these LC addresses are set minus ("1") or plus ("0"). The sign bits of specific LC addresses are then checked during the execution of the routines and, depending on whether they are plus or minus, special series of program instructions are executed.

(c) Block 3. The following discrete outputs are set plus: start computation, computer running, second stage engine cutoff, atuopilot scale factor, AGE data clock, and time reference system gate.

(d) Block 4. The processor real time count is read for utilization by the individual routines.

(e) Block 5. The accelerometer subroutine is executed to verify that the X, Y, and Z velocity signals from the accelerometers equal zero.

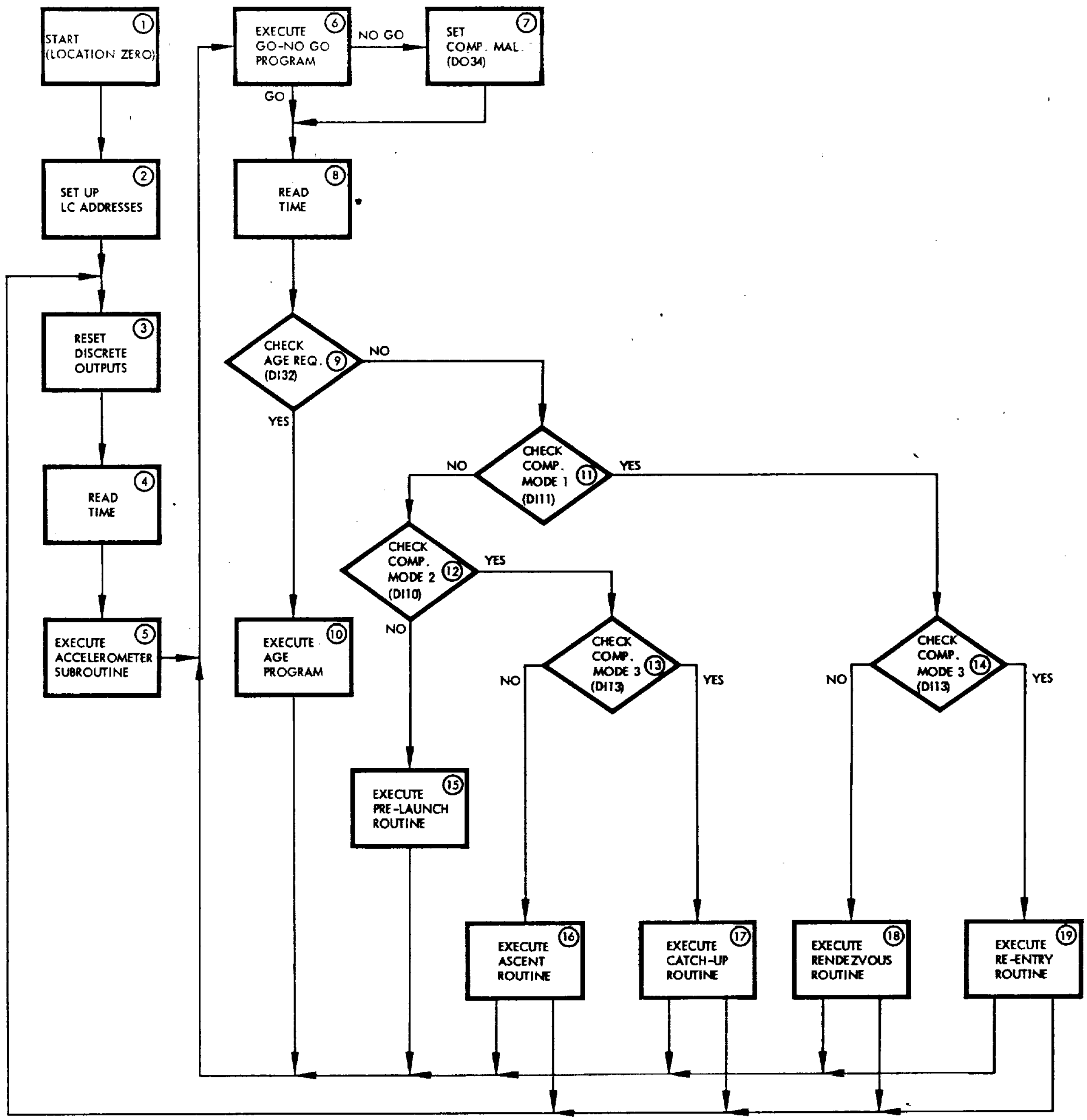


Figure 8-27 Executor Routine Program Flow

(f) Block 6. A special go, no-go diagnostic program is executed to determine if the basic computer arithmetic circuits are functioning properly. If these circuits fail, the NO GO path is followed; if there is no failure, the GO path is followed.

(g) Block 7. Program instruction PRO34 is executed. The execution of this instruction causes the computer malfunction circuit to be conditioned.

(h) Block 8. The processor real time count is read and updated for utilization by the individual routines.

(i) Block 9. Program instruction CLD32 is executed to determine the condition of the AGE request discrete input. If the input is a "1," the YES path is followed; if the input is a "0," the NO path is followed.

(j) Block 10. Special check-out tests are executed by the AGE. Both the Gemini Launch Vehicle and the computer can be checked out.

(k) Blocks 11 through 14. Program instructions CLD10, CLD11, and CLD13 determine the condition of the discrete inputs from the Computer Mode switch. This switch is manually controlled by the pilot and, depending upon which mode is selected, causes a particular routine to be executed until the switch setting is changed or until the computer is turned off. The combinations of Computer Mode switch discrete inputs required to select a particular routine are as follows:

Routine	Discrete Inputs		
	DI10	DI11	DI13
Pre-Launch	"0"	"0"	"0"
Ascent	"1"	"0"	"0"
Catch-Up	"1"	"0"	"1"
Rendezvous	"0"	"1"	"0"
Re-Entry	"0"	"1"	"1"

(1) Blocks 15 through 19. Depending on the setting of the Computer Mode switch, one of these operational routines is selected. The individual routines are discussed in subsequent paragraphs.

Pre-Launch Routine

The Pre-Launch routine provides the instructions required to check out the computer prior to launch and to read in special data for future use. This routine performs sum-checks on all sectors within the computer memory. These checks are performed by adding the contents of all memory addresses within a sector and comparing the sum with a pre-stored constant. If the constant and the sum are not equal, the computer malfunction latch is set by program instruction PRO34. If the sum check is successful, special data is stored in pre-determined memory addresses by the common subroutines. These subroutines are discussed in later paragraphs.

Ascent Routine

The Ascent routine provides the computations required for back-up ascent guidance. After the computer has been placed in the Ascent mode, special data is transferred to the computer via the Digital Command System. This data is then continually updated and used to keep track of the orbit plane and the platform attitude with respect to Earth. Thirty seconds after the special data is first transferred to the computer, the Inertial Guidance System is placed in the Inertial mode. The computer continually monitors and stores the platform gimbal angle values during this time. After lift off, the computer performs a back-up guidance function. If necessary, however, the computer can be used to perform primary guidance during Ascent.

Catch-Up Routine

The Catch-Up routine is not utilized in S/C 3, 4 or 7, because they are of non-rendezvous configuration. For information pertaining to this routine, refer to Vol. II of this document.

Rendezvous Routine

The Rendezvous routine is not utilized in S/C 3, 4 or 7, because they are of non-rendezvous configuration. For information pertaining to this routine, refer to Vol. II of this document.

Re-Entry Routine

The Re-Entry routine provides the computations required for re-entry guidance. During the Re-Entry mode, the retro velocity is monitored and retro velocity errors are calculated. The distance and heading of the spacecraft with respect to the desired landing site are calculated, and the down range travel to touchdown is predicted. The routine also provides signals to command the spacecraft roll maneuvers during re entry and provides a display of attitude errors as detailed on pages 8-137 and 8-138.

NOTE

The following subroutines are common to the previously described routines: Gimbal Angle, Accelerometer, Digital Command System, Instrumentation System, and Manual Data. Therefore, a description of each of these subroutines follows.

Gimbal Angle Subroutine

The Gimbal Angle subroutine reads and processes the gimbal angles for the pitch, yaw, and roll axes of the Inertial Platform. During a computer word time, the gimbal angle processor reads in one gimbal angle value and transfers a previously read gimbal angle value to the accumulator. This method enables a faster processing operation than if the angle for each axis were processed individually. Approximately 5 ms elapses between the processing of one gimbal angle value and the processing of the next gimbal angle value. (The gimbal angle value is the binary equivalent of the actual gimbal angle.)

Accelerometer Subroutine

The Accelerometer subroutine processes velocity signal inputs from the Inertial Measuring Unit. These signals, which represent velocity for the X, Y, and Z axes of the spacecraft, are generated by accelerometers. Due to the construction and adjustment of the accelerometers, the signals contain inherent bias and alignment errors. The subroutine corrects these errors and stores the corrected velocity values in predetermined computer memory locations. The computer input processor reads the X, Y, and Z velocity signals, and transfers them to the processor delay line. The delay line is then read by the subroutine at periodic intervals which depend on the selected mode or routine.

Digital Command System Subroutine

The Digital Command System subroutine reads and processes data furnished by the Digital Command System (DCS). The DCS furnishes the computer with special 24-bit words consisting of 6 address bits and 18 data bits. The address bits indicate where the data bits are to be stored in the computer memory. The subroutine first determines if data is available from the DCS. If data is available, the subroutine then reads the data into the accumulator. Next, the address and data bits are separated. The data bits are then stored in the computer memory address specified by the address bits. After this data is stored, it is used as constants by other subroutines. The DCS subroutine also contains instructions which provide extended DCS addresses. (Address 100-117). The recognition of addresses 20 and 21 excersises the proper operational program loops to store the data in the computer. For each DCS extended address insert, it is necessary to make two transmissions and this must be accomplished in the proper order (i.e. - DCS address 20 first, 21 next). On the first cycle through the DCS subroutine, address 20 is recognized and the associated data is stored as high order data. On the second cycle, address 21 is recognized and the associated data yields low order data plus the DCS extended address word. With the DCS extended address, it is possible to insert 26 - bit words into the computer.

Instrumentation System Subroutine

The Instrumentation System subroutine assembles special data and transfers it to the Instrumentation System (IS). Every 2.4 seconds, 21 data words are transferred to the IS by the subroutine. The transferred data words are the stored results of other subroutines. The types of data words transferred include velocity changes for the X, Y, and Z axes, gimbal angle values for the pitch, roll, and yaw axes, and radar range. Once every 2.4 seconds, the IS sync discrete input occurs. When the input occurs, the data words to be transferred are assembled in a special IS memory buffer. The buffer consists of 21 predetermined memory addresses. A special memory address is used as a word selection counter to determine which data words in the IS memory buffer are to be transferred to the IS.

Manual Data Subroutine

The Manual Data subroutine determines when data is transferred from the Manual Data Keyboard (MDK) to the computer and from the computer to the Manual Data Readout (MDR). The subroutine consists of approximately 1000 instructions which are used to govern the generation of signals that control circuit operation in the MDK and MDR.

Interfaces

Figure 8-28 shows the equipment which interfaces with the computer. The diagram also contains references to the individual equipment interface diagrams.

Inertial Platform (Figure 8-29)

The computer supplies 400 cps excitation to the rotors of three resolvers located on the pitch, roll, and yaw gimbal axes of the Inertial Platform. Movement of the rotors of any of these resolvers away from their zero (platform-caged) reference causes the output voltage of the stator winding to be phase-shifted relative to the reference 400 cps voltage inputs to the computer: a reference voltage from the compensator winding (pitch, yaw, and roll references), and a phase-shifted voltage from the stator winding (pitch, yaw, and roll gimbal angles).

The following PRO instruction programming is associated with the Inertial Platform interface:

Signal	Address X	Address Y
Read pitch gimbal	6	3
Read roll gimbal	6	4
Read yaw gimbal	6	5

The gimbal angles are read no sooner than 5 ms from each other, and the total reading time for all three angles is no greater than 30 ms. The angles are read once per computation in the Re-Entry mode, and once every 50 ms in the

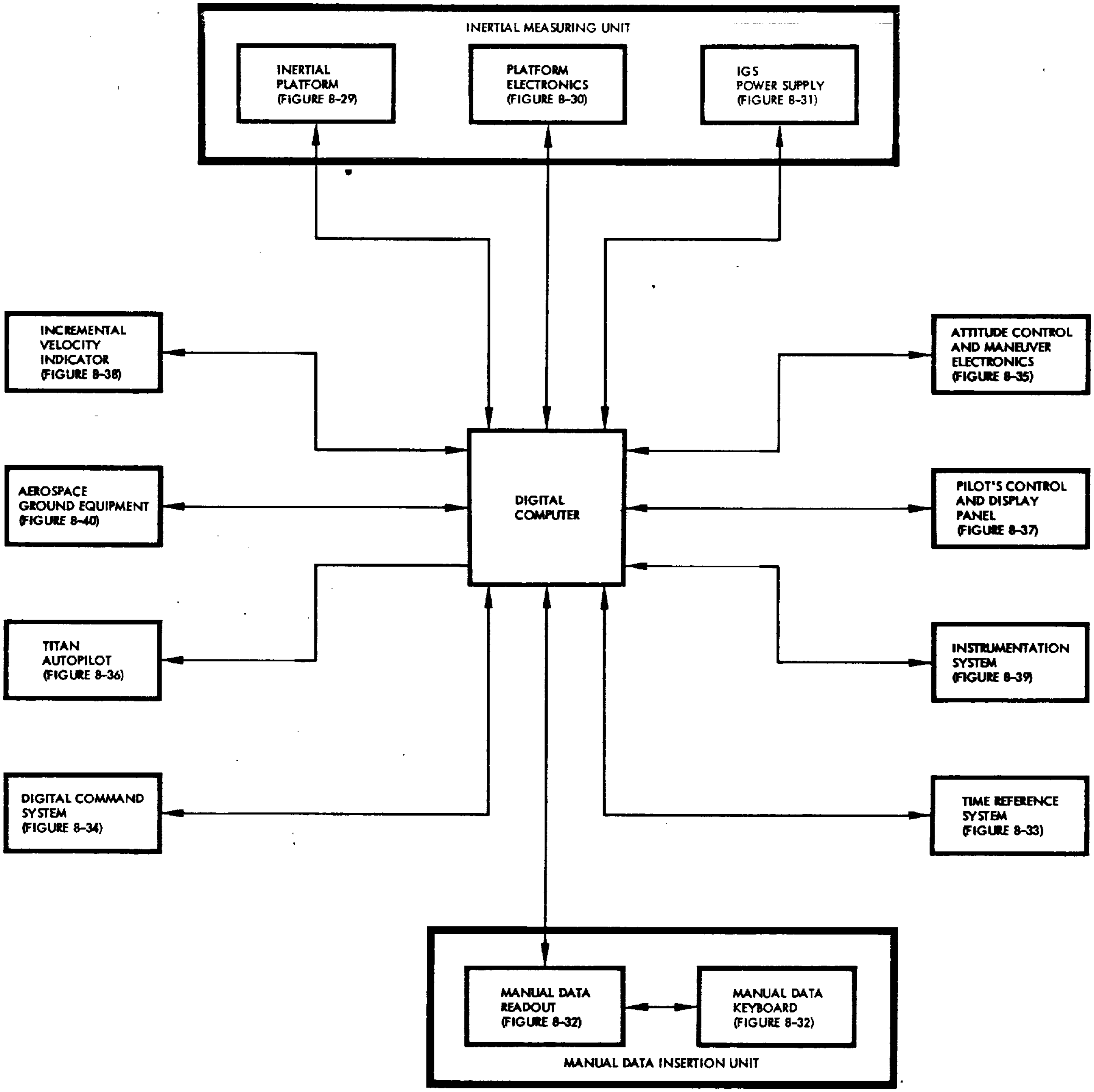


Figure 8-28 Computer Interfaces

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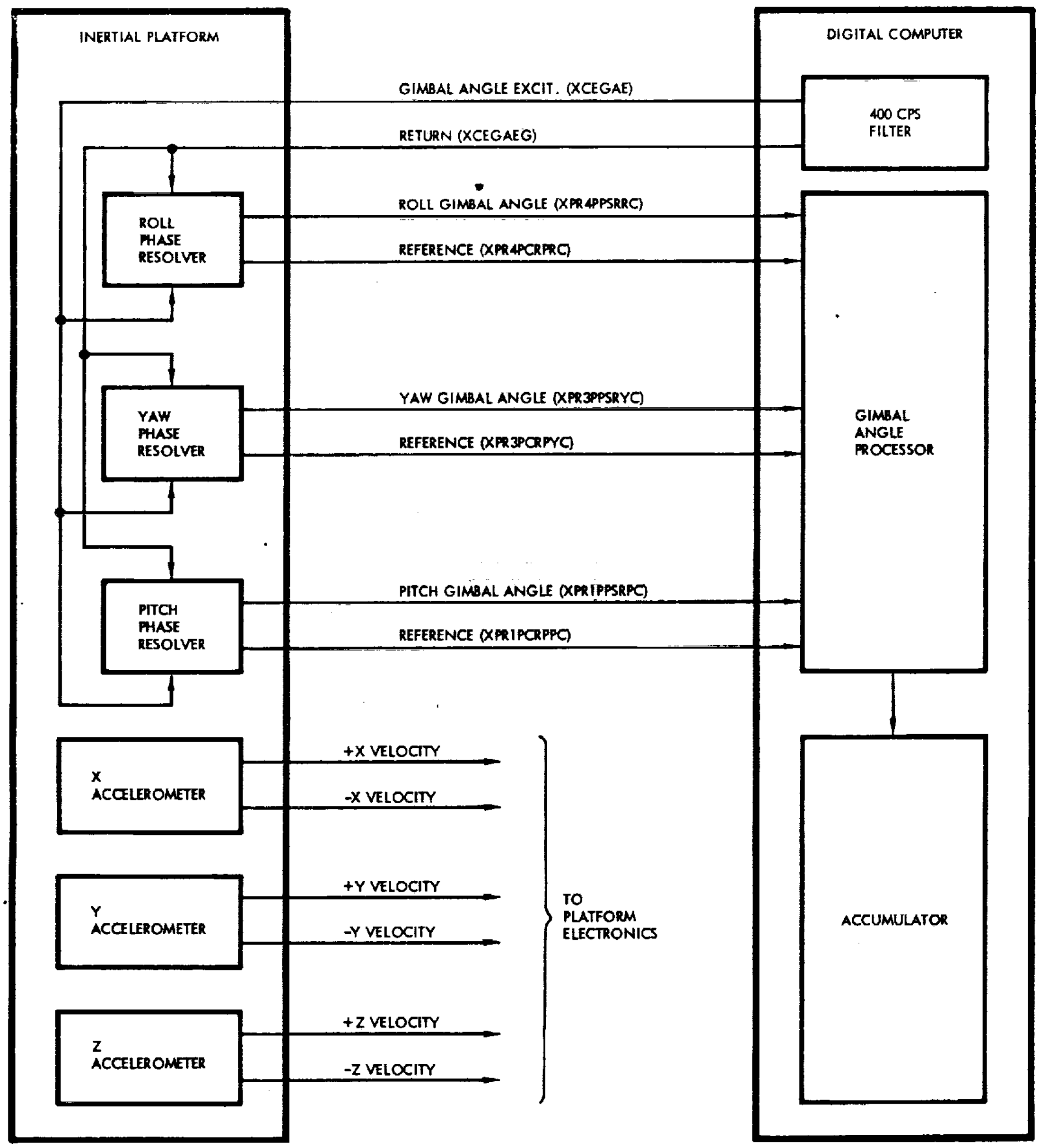

FM2-8-29

Figure 8-29 Computer-Platform Interface

Ascent mode. These angles are gated, as true magnitude, into the accumulator S, and 1 through 14 bit positions with the 15 through 25 bit positions being zero. The accumulator value from the first PRO instruction is discarded. Each of the next three PRO instructions results in an accumulator value of the gimbal angle read by the previous PRO instruction, as follows:

(a) PRO36 (read pitch; process previously read angle)
(b) Discard previously read angle
(c) Wait 5 ms
(d) PRO46 (read roll; process pitch)
(e) STO pitch
(f) Wait 5 ms
(g) PRO56 (read yaw; process roll)
(h) STO roll
(i) Wait 5 ms
(j) PRO36 (read pitch; process yaw)
(k) STO yaw

The computer inputs from the Inertial Platform are summarized as follows:

(a) Roll gimbal angle (XPR4PPSRRC) and reference (XPR4PCRPRC)
(b) Yaw gimbal angle (XPR3PPSRYC) and reference (XPR3PCRPYC)
(c) Pitch gimbal angle (XPR1PPSRPC) and reference (XPR1PCRPPC)

The computer output to the Inertial Platform is summarized as follows:

Gimbal angle excitation (XCEGAE) and return (XCEGAEG)

Platform Electronics (Figure 8-30)

Outputs derived from each of the three platform accelerometers are supplied to the computer as incremental velocity pulses (+X and -X delta velocity, +Y and -Y delta velocity, and +Z and -Z delta velocity). An up level on one line denotes a positive increment of velocity while an up level on the other line denotes a negative increment of velocity.

The following PRO instruction programming is associated with the Platform Electronics interface:

Signal	Address X	Address Y	Processor Phase Time
Read X delta velocity	5	4	2
Read Y delta velocity	5	4	3
Read Z delta velocity	5	4	4

The input processor accumulates the incremental velocity pulses on the processor delay line in two's-complement form. The velocity pulses have a maximum frequency of 3.6 kc per channel with a minimum separation of 135 usec between any plus and minus pulse for a given axis. Three input circuits are used to buffer the plus and minus pulses, one circuit for each axis. The buffered velocity pulse inputs are sampled during successive processor phases and read into a control circuit. This control circuit synchronizes the inputs with the processor timing and establishes an add, subtract, or zero control for the processor carry-borrow circuit. The accumulated velocity quantities are read into the accumulator

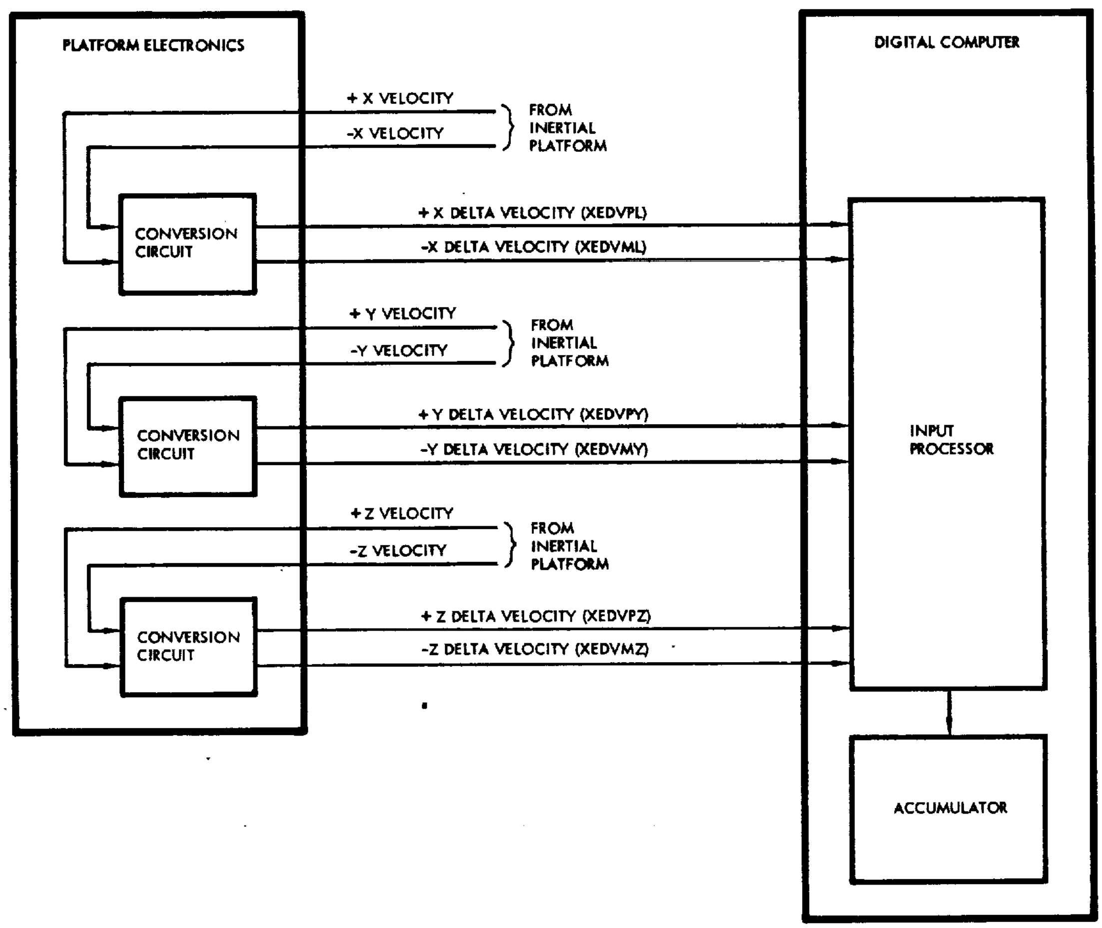

Figure 8-30 Computer-Platform Electronics Interface

S, and 1 through 12 bit positions in two's-complement form via a single PRO45 instruction, as follows:

(a) Processor phase 2 - read accumulated X velocity

(b) Processor phase 3 - read accumulated Y velocity

(c) Processor phase 4 - read accumulated Z velocity

As the accelerometer values are read into the accumulator, the delay line is automatically zeroed so that each reading represents the change in velocity from the previous reading.

The computer inputs from the Platform Electronics are summarized as follows:

(a) +X delta velocity (XEDVPL)

(b) -X delta velocity (XEDVML)

(c) +Y delta velocity (XEDVPY)

(d) -Y delta velocity (XEDVMY)

(e) +Z delta velocity (XEDVPZ)

(f) -Z delta velocity (XEDVMZ)

IGS Power Supply (Figure 8-31): The computer supplies a filtered 28 VDC signal to the IGS Power Supply to control the DC power supplied to the computer. The IGS Power Supply supplies power to the computer within 0.3 second after receiving the 28 VDC power control signal. When the computer power control signal drops to 2 VDC, the IGS Power Supply removes DC power from the computer within 0.3 second. The 26 VAC, 400 cps power furnished to the computer by the IGS Power Supply is not controlled by the computer power control signal, and is therefore present at the computer whenever the IGS Power Supply is operating.

The computer inputs from the IGS Power Supply are summarized as follows:

(a) +27.2 VDC (XSP27VDC) and return (XSP27VDCRT)

(b) -27.2 VDC (XSM27VDC) and return (XSM27VDCRT)

(c) -20 VDC (XSP20VDC) and return (XSP20VDCRT)

(d) +9.3 VDC (XSP9VDC) and return (XSP9VDCRT)

(e) 26 VAC (XS26VAC) and return (XS26VACRT)

(f) +28 VDC filtered (XSP28VDC) and return (XSP28VDCRT)

The computer output to the IGS Power Supply is summarized as follows:

Power control (XCEP)

Auxiliary Computer Power Unit (ACPU) (Figure 8-31)

The ACPU functions in conjunction with the IGS Power Supply to buffer power interruptions and depressions. When the ACPU senses a power interruption or depression, it supplies the power loss sensing signal to the power sequencing circuits in the computer. These circuits then maintain the computer power constant until the power interruption or depression ends (up to a maximum of 100 m sec).

The computer output to the ACPU is summarized as follows:

Power Control (XCEP)

The computer input from the ACPU is summarized as follows:

Power loss sensing (XQBND)

+28 VDC Filtered (XSP28VDC)

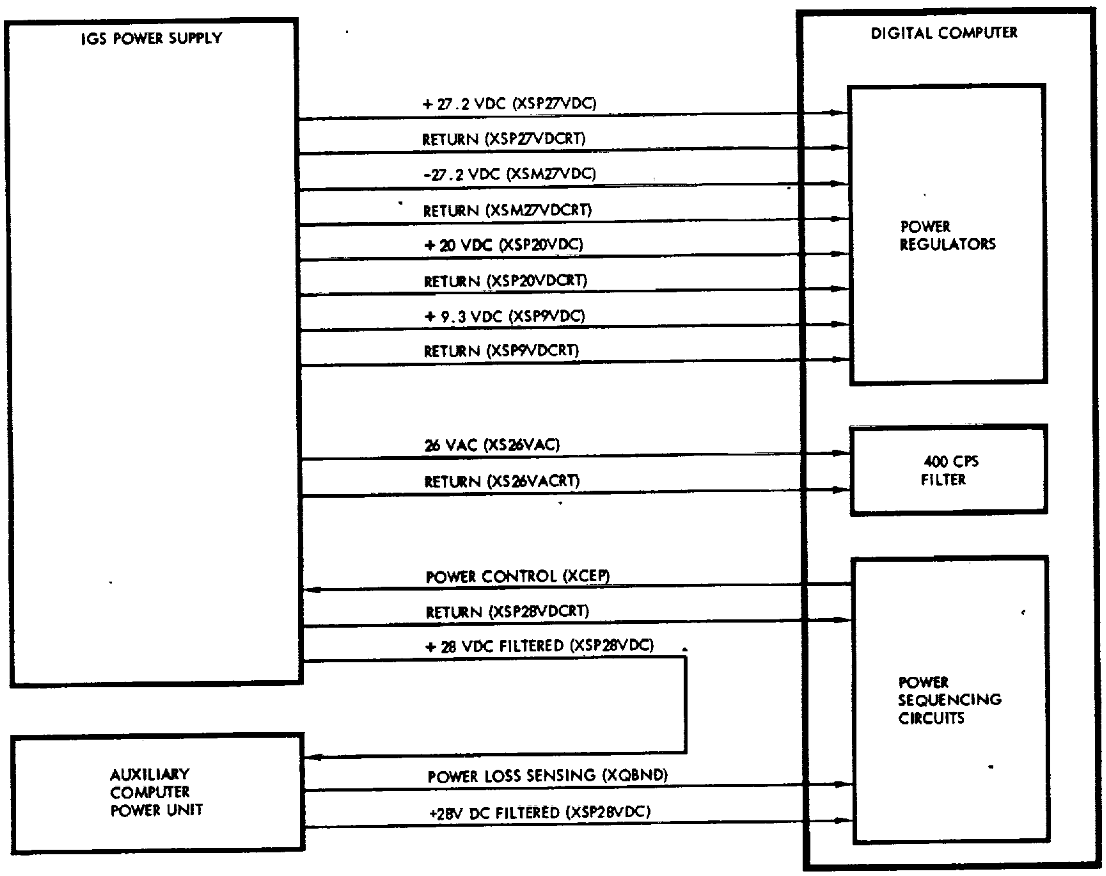

Figure 8-31 Computer-Power Supply Interface

Manual Data Insertion Unit (MDIU) (Figure 8-32)

The MDIU can insert into, and/or read out of, the computer up to 99 data words. It provides the crew with a means of updating certain data stored in the computer by inserting new data into the appropriate memory location. It also provides a capability to verify the data stored in a number of additional memory locations. Two of the quantities which may be inserted (T_R and T_X) are transferred to the Time Reference System by the computer, following insertion.

The MDIU consists of two units: The Manual Data Keyboard (MDK) and the Manual Data Readout (MDR). The MDK has a keyboard containing 10 push-button switches used during data insertion and readout. To insert data, the pilot always depresses seven Data Insert push-button switches; the first two set up the address of the computer memory location in which data is to be stored, and the last five set up the actual data. Each digit inserted is also displayed for verification. Following the insertion and verification of the seventh digit, the ENTER push-button switch is pressed to store the data in the selected memory location. If verification of any digit cannot be made, the CLEAR push-button switch is pressed and the address and data must be set up again. The MDR sequentially displays for verification the digits inserted by the pilot. This unit can also be used to recheck quantities stored in the computer memory. This operation is accomplished by inserting and verifying only the first two (address) digits and then depressing the READ OUT push-button switch. The selected data is then displayed for verification. If the pilot attempts to insert data in an invalid address, attempts to read data out of an invalid address, inserts more than seven digits, or fails to insert a two-digit address prior to depressing the

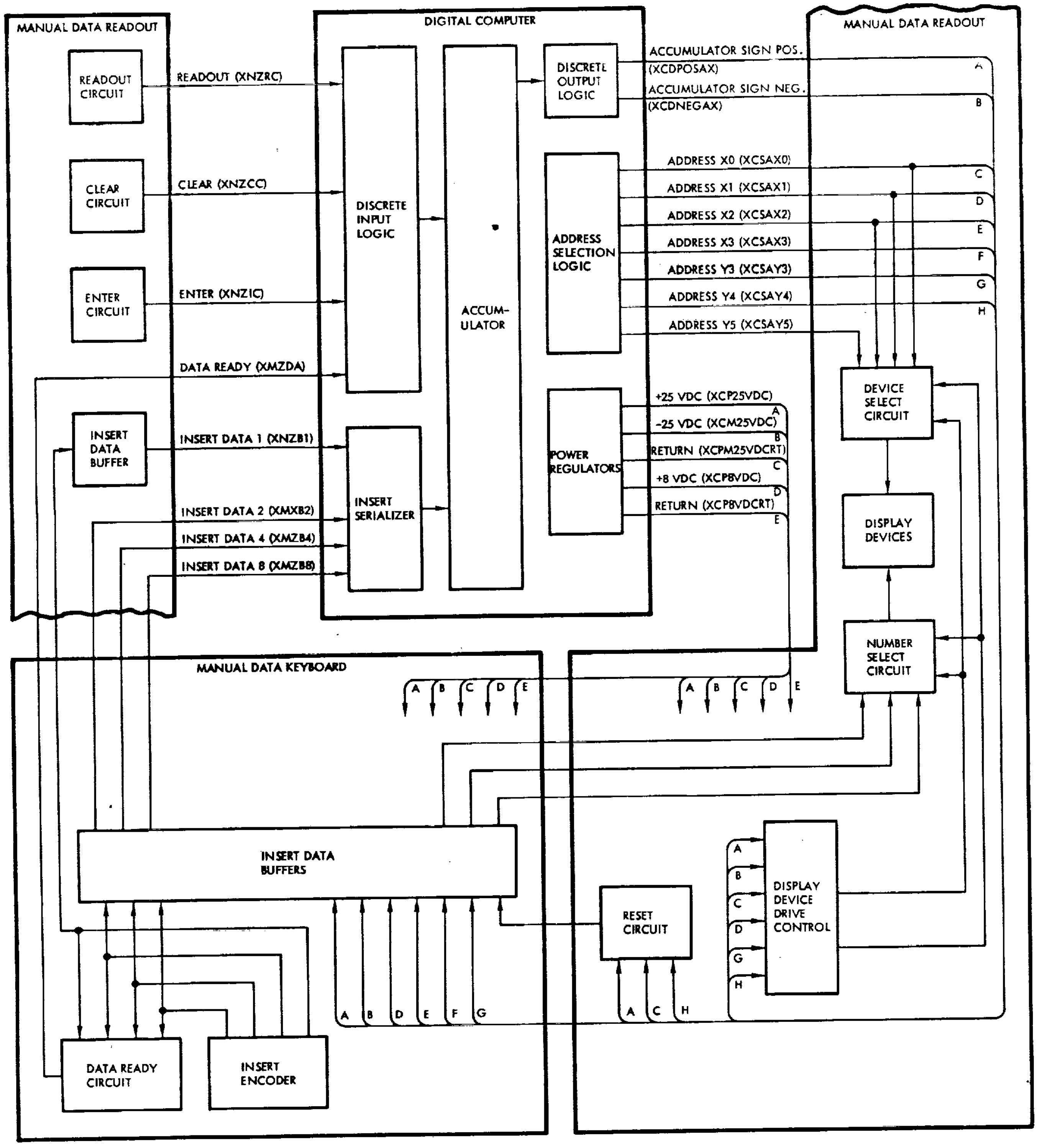

Figure 8-32 Computer-MDIU Interface

ENTER or READ OUT push-button switch, the seven digits displayed are all zeros indicating a pilot error.

The following CLD instruction programming is associated with the MDIU interface:

Signal	Address X	Address Y
Data ready	1	0
Enter	2	0
Readout	3	0
Clear	4	0

The following PRO instruction programming is associated with the MDIU interface:

Signal	Address X	Address Y
Digit magnitude weight 1	0	3
Digit magnitude weight 2	1	3
Digit magnitude weight 4	2	3
Digit magnitude weight 8	3	3
Reset KI01, DI02, and DI03	0	4
Display device drive	1	4
Digit select weight 1	0	5
Digit select weight 2	1	5
Digit select weight 2	1	5
Digit select weight 4	2	5
Read MDIU insert data	3	4

The pilot must depress the CLEAR push-button switch for the first quantity to be inserted or displayed. Upon the recognition of DI04 on, the program sets DO40 off. This results in resetting DI01, DI02, and DI03, and clearing the MDIU buffer. The program then sets DO41 off to reset the display drivers.

When a digit push-button switch is depressed, the binary coded decimal (BCD) code is entered into the buffer and DI01 is turned on. The program reads the buffer into accumulator bit positions 1 through 4 and sets DO40 off. Following this, the program sends out a code by means of DO50, DO51, and DO52 to select the digit to be displayed. The program then sets DO41 on to turn on the display drivers, and sends a BCD digit to the buffer by means of DO30, DO31, DO32, and DO33. The program waits 0.5 second and sets DO40 and DO41 off. The astronaut must wait until the digit is displayed before entering the next digit. After all seven digits have been entered and displayed, the pilot depresses the ENTER push-button switch. This results in DI02 being set on. The program then sets DO40 off, and converts the five data digits to binary. This data is scaled and stored in memory according to the two-digit address.

To read data out of the computer, the pilot enters the two-digit address of the quantity to be displayed and then depresses the READ OUT push-button switch. This results in DI03 being set on. The computer then sets DO40 off, converts the requested quantity to BCD, and sends the BCD data to the display buffer one digit at a time in 0.5-second intervals.

The computer inputs from the MDIU are summarized as follows:

(a) Readout (XMZRC) - The up level of this signal denotes that the two previously inserted digits are to be used as the address of a quantity to be displayed.

(b) Clear (XNZCC) - The up level of this signal denotes that the previously inserted digits are incorrect and the insert sequence must be repeated.

(c) Enter (XNZIC) - The up level of this signal denotes that the previously inserted digits have been verified and should be stored in the computer memory.

(d) Data ready (XMZDA) - The up level of this signal denotes that a digit has been inserted. The computer samples this line at least 20 times per second to allow continuous insertion of data.

(e) Insert data 1, 2, 4, and 8 (XNZB1, XMZB2, XMZB4, and XMZB8) - These four signals, denoting one BCD character, are supplied to the computer for each decimal digit inserted.

The computer outputs to the MDIU are summarized as follows:

(a) Accumulator sign positive (XCDPOSAX) - The up level of this signal on a set input causes the addressed latch to be set.

(b) Accumulator sign negative (XCDNEGAX) - The up level of this singal on a reset input causes the addressed latch to be reset.

(c) Addressing - Seven lines provide the capability of addressing all latches in the MDIU. The following X and Y address lines are provided:

(1) MDIU address X0 (XCSAX0)

(2) MDIU address X1 (XCSAX1)

(3) MDIU address X2 (XCSAX2)

(4) MDIU address X3 (XCSAX3)

(5) MDIU address Y3 (XCSAY3)

(6) MDIU address Y4 (XCSAY4)

(7) MDIU address Y5 (XCSAY5)

By selecting one X and one Y address line at a time, a total of 12 addresses can be formed.

(d) DC power - Regulated DC power is supplied to the MDIU as follows:

(1) +25 VDC (XCP25VDC)
(2) -25 VDC (XCM25VDC)
and return (XCPM25VDCRT)

(3) +8 VDC (XCP8VDC) and return (XCP8VDCRT)

Time Reference System (TRS) (Figure 8-33)

The TRS counts elapsed time ET from lift-off through impact, counts down time to retrograde (T_R) on command, and counts down time to equipment reset (T_X) on command, all in 1/8-second increments. The computer receives T_R and T_X data words from the MDIU and automatically transfers them to the TRS. When the computer receives a display request from the MDIU for T_R, or when the computer

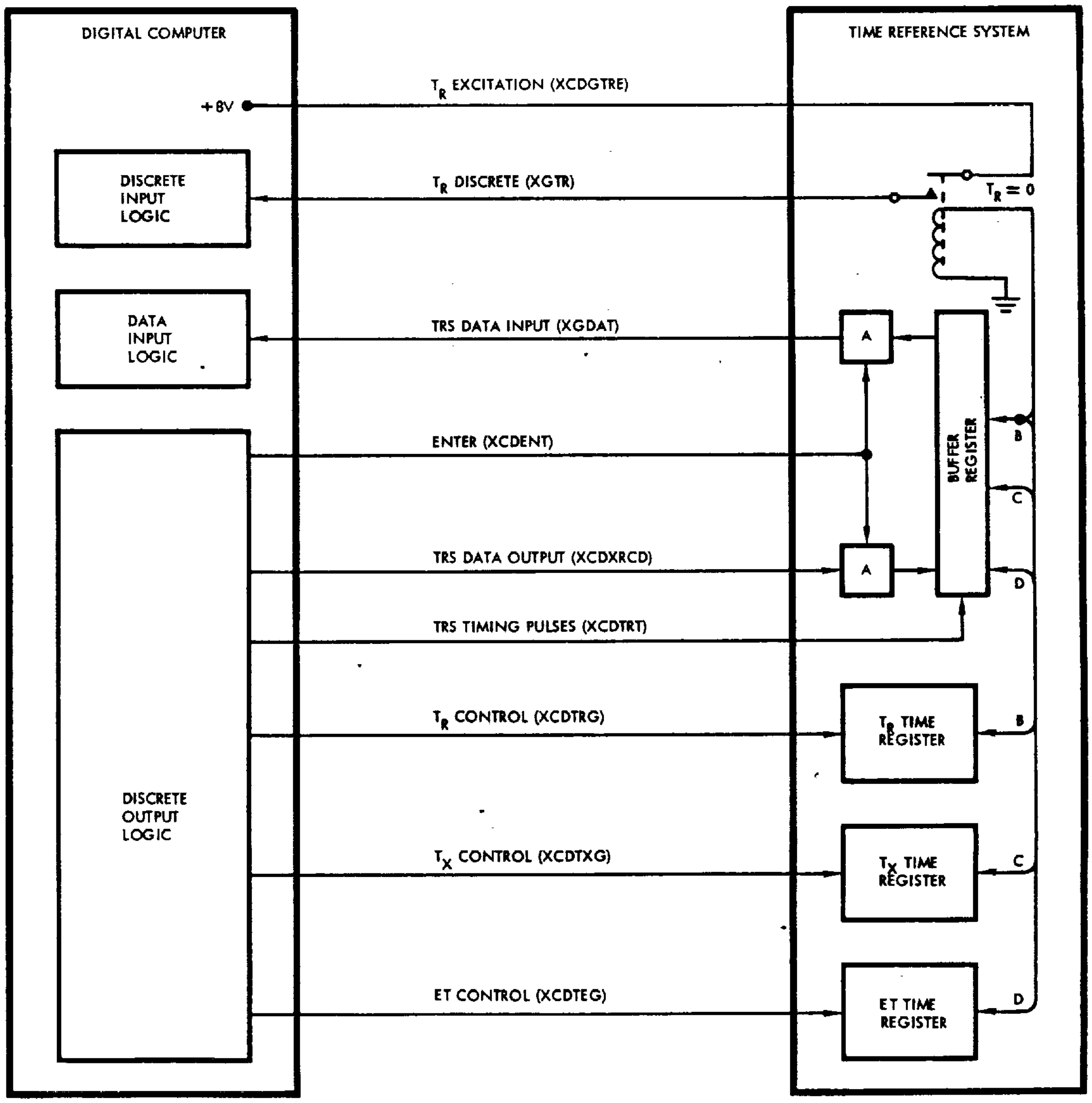

Figure 8-33 Computer-TRS Interface

program requires ET, the TRS transfers them to the computer.

The following CLD instruction programming is associated with the TRS interface.

Signal	Address X	Address Y
T_R discrete	5	0

The following PRO instruction programming is associated with the TRS interface:

Signal	Address X	Address Y
ET control	4	1
T_X control	5	2
T_R control	5	6
Enter	1	2
TRS data and timing pulses	0	2
TRS control reset	4	1

In the readout mode, the computer transfers T_R or T_X data words to the TRS. The mode is initiated by setting D021 on. The 24 bits of data to be sent to the TRS are then placed in the accumulator by 24 consecutive sets of PRO20 and SHR1 (shift right one place) instructions. With each PRO instruction, a timing pulse is automatically initiated 70 usec after the beginning of the data pulse. The timing pulse is terminated so that its up level is 139 usec. After bit 24 has been sent to the TRS, the program generates one of two control gates (T_R, or T_X).

Between 9 and 15 ms later, the computer terminates the TRS control gate.

The enter mode is initiated by setting D021 off. One of two control gates (ET, or T_R) is generated by the program and terminated between 9 and 15 ms later. After termination of the control gate, the program enters a subroutine consisting of 25 consecutive sets of PRO10 and SHR1 instructions. Every time a PRO operation is called for, a timing pulse is generated by the same logic as in the readout mode. The timing pulse is sent to the TRS to cause the addressed data to be supplied to the computer. The first bit received is discarded with the final SHR1 instruction. The second bit received is the least significant bit and is shifted into accumulator bit position 25 at the completion of the twenty-fifth set of PRO20 and SHR1 instructions. When T_R equals zero, a relay in the TRS shorts the T_R excitation line to the T_R discrete line. The T_R discrete signal then causes the computer to start re-entry calculations.

The computer inputs from the TRS are summarized as follows:

(a) T_R discrete (XGTR) - The up level of this signal signifies that the computer should begin re-entry calculations.

(b) TRS data input (XGDAT) - All data transfers from the TRS to the computer occur on this line. The data word on the line is determined by which control gate the computer actuates prior to the actual data transfer. The up level is a binary "1."

The computer outputs to the TRS are summarized as follows:

(a) T_R excitation (XCDGTRE) - The computer supplies +8 VDC through a resistor to the TRS as the T_R excitation input. When T_R equals zero, the T_R relay causes the T_R excitation input to be transferred to the computer as the T_R discrete signal.

(b) Enter (XCDENT) - The up level of this signal signifies that data is to be transferred from the TRS to the computer when the transfer clocks occur. The down level signifies that data is to be transferred from the computer to the TRS.

(c) TRS data output (XCDXRCD) - All data transfers from the computer to the TRS occur on this line. The data word on the line is determined by which control gate (T_R, or T_X) the computer has actuated. The up level is a binary "1."

(d) TRS timing pulses (XCDTRT) - These 3.57 kc timing pulses cause the computer data to be shifted into or out of the TRS buffer register for transfer to or from the computer.

(e) T_R control (XCDTRG) - The up level of this signal causes the transfer of data between the TRS buffer register and the TRS T_R register. The direction of transfer is determined by the level of the enter signal.

(f) T_X control (XCDTXG) - The up level of this signal causes the transfer of data between the TRS buffer register and the TRS T_X register. The direction of transfer is determined by the level of the enter signal.

(g) ET control (XCDTEG) - The up level of this signal causes the transfer of data between the TRS buffer register and the TRS ET register. The direction of transfer is determined by the level of the enter signal.

Digital Command System (DCS) (Figure 8-34)

The DCS accepts BCD messages from the ground stations at a 1 kc rate, decodes the messages, and routes the data to either the TRS or the computer. In addition, the DCS can generate up to 64 discrete commands.

Signal	Address X	Address Y
DCS ready	6	0

The following PRO instruction programming is associated with the DCS interface:

Signal	Address X	Address Y
Computer ready	1	0
DCS shift pulse gate	0	0

When data is to be sent to the computer, the DCS supplies the computer with a DCS ready discrete input (DI06). This input is sampled every 50 ms or less in all computer modes except during the 1/8-second interval in the Ascent mode when reading ET at lift-off. To receive DCS data, the computer supplies a series of 24 DCS shift pulses at a 500 kc repetition rate by setting DO01 off and programming a PR00 instruction. These shift pulses cause the data contained in

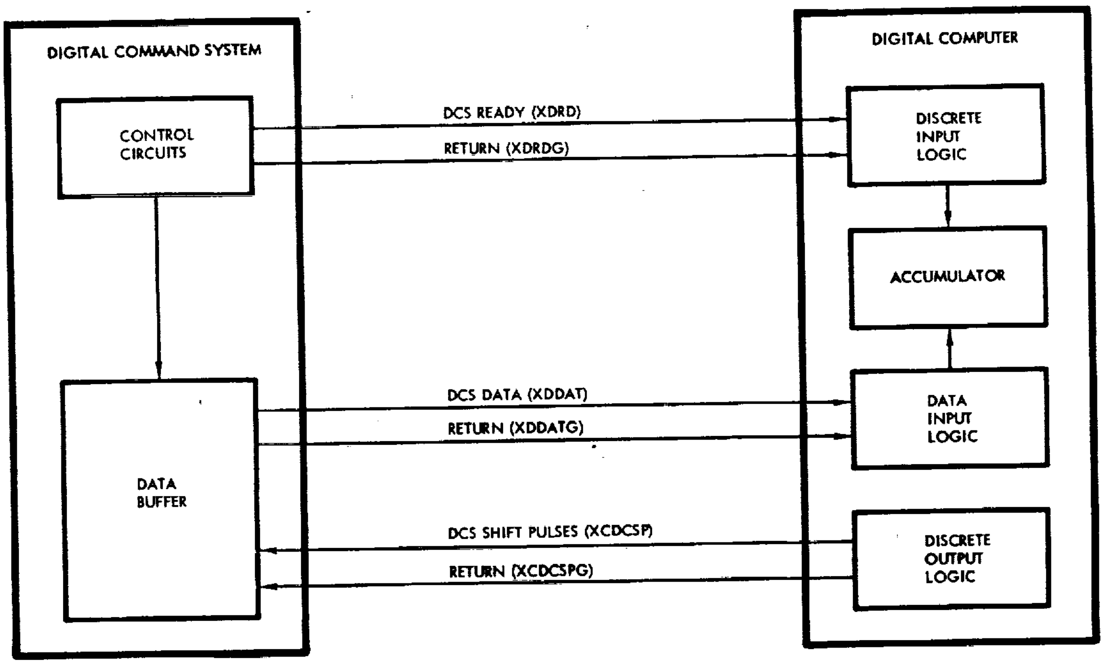

Figure 8-34 Computer-DCS Interface

the DCS buffer register to be shifted out on the DCS data line and read into accumulator bit positions 1 through 24, with positions 19 through 24 containing the assigned address of the associated quantity and positions 1 through 18 containing the quantity. Bit position 19 (address portion) and bit position 1 (data portion) are the most significant bits.

The computer inputs from the DCS are summarized as follows:

(a) DCS ready (XDRD) and return (XDRDG) - The down level of this signal signifies that the DCS is ready to transfer data to the computer.

(b) DCS data (XDDAT) and return (XDDATG) - This serial data from the DCS consists of 24 bits, with 6 being address bits and 18 being data bits.

The computer output to the DCS is summarized as follows:

DCS shift pulses (XCDCSP) and return (XCDCSPG) - The computer supplies these 24 shift pulses to the DCS to transfer data contained in the DCS buffer register out on the DCS data line.

Rendezvous Radar

The Rendezvous Radar is not installed in S/C 3, 4 or 7. For information pertaining to this system, refer to Vol. II of this document.

Attitude Display/Attitude Control and Maneuver Electronics (ACME) (Figure 8-35)
During the Ascent mode, the computer generates pitch, roll, and yaw attitude error signals and supplies them to the Attitude Display. The pilot utilizes the Attitude Display to monitor the performance of the ascent guidance equipment.

During the Re-Entry mode, the computer generates a roll attitude error or bank rate signal and supplies it to the Attitude Display and the ACME. If range to touchdown with zero lift is equal to, or greater than, the computed range to the desired touchdown point, a bank rate command equivalent to a 20 degree per second roll rate is provided on the roll attitude error output line. Also, during the Re-Entry mode, the computer generates cross range and down range error signals and supplies them to the Attitude Display for the pilots' use in manually controlling the re-entry flight path of the spacecraft.

The following PRO instruction programming is associated with the Attitude Display and ACME interfaces:

Signal	Address X	Address Y
Pitch error command	7	0
Yaw error command	7	1
Roll error command	7	2
Pitch resolution	2	0
Yaw resolution	3	0
Roll resolution	4	0

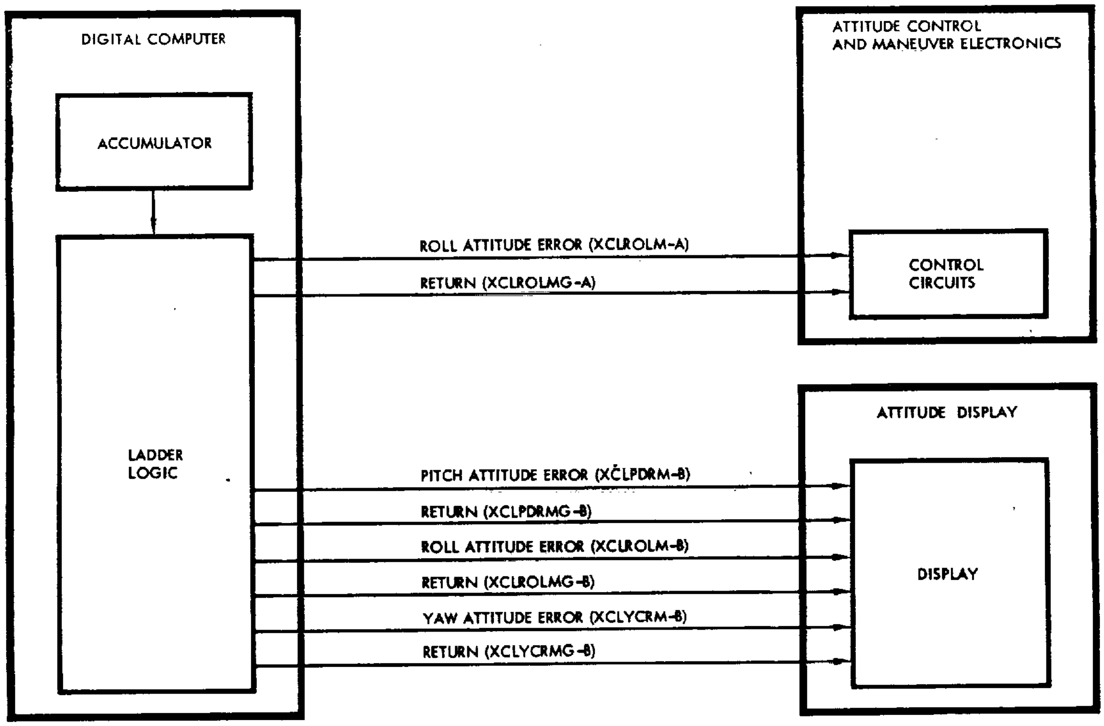

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Figure 8-35 Computer-Attitude Display/ACME Interface

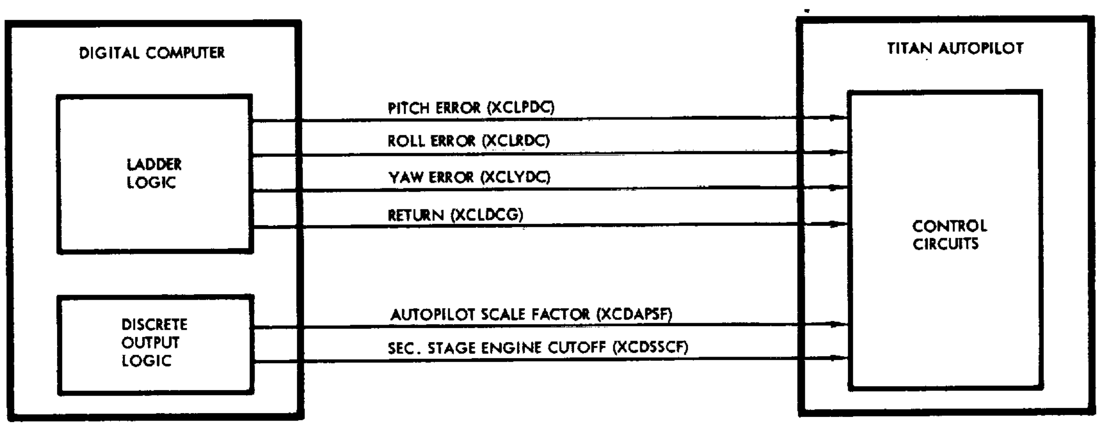

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Figure 8-36 Computer-Autopilot Interface

The pitch, yaw, and roll error commands are written into a seven-bit register from accumulator bit positions S, and 8 through 13, with a PRO instruction having an X address of 7. The outputs of the register are connected to ladder decoding networks which generate a DC voltage equivalent to the buffered digital error. This analog voltage is then sampled by one of three sample and hold circuits; while one circuit is sampling the ladder output, the other two circuits are holding their previously sampled value. The minimum sample time is 2 ms, and the maximum hold time is 48 ms. The Y address of the previously mentioned PRO instruction selects the one sample and hold circuit that is to sample the ladder output. The output of each sample and hold circuit is fed into an individual ladder amplifier where the DC analog voltage for each channel is made available for interfacing with the Titan Autopilot.

The DC analog outputs are also fed through individual range switches and magnetic modulators where the DC voltages are converted to 400-cycle analog voltages. The range switches, which are controlled by means of discrete outputs, can attenuate the DC voltages being fed into the magnetic modulators by a factor of 6-to-1. The addressing of the discrete outputs for controlling the range switches is as follows:

(a) Pitch or down range error (D002) -
(b) Yaw or cross range error (D003) -
(c) Roll error (D004) -

} plus for low range; minus for high range.

The error commands are written every 50 ms or less. The updating period, however, is dependent upon the computer mode of operation. For the Re-Entry mode (and the orbital insertion phase of ascent guidance), the error commands are updated once per computation cycle or every 0.5 second or less. For first and second stage ascent guidance, the error commands are updated every 50 ms or less.

The computer outputs to the Attitude Display and ACME are summarized as follows:

(a) Pitch attitude error (XCLPDRM) and return (XCLPDRMG) - Two identical sets of outputs (A and B) are time-shared between pitch attitude error (during Ascent) and down range error (during Re-Entry).

  (1) Pitch attitude error (Ascent) to Attitude Display

  (2) Down Range error (Re-Entry) to Attitude Display

(b) Roll attitude error/bank rate command (XCLROLM) and return (XCLROLMG) - Two identical sets of outputs (A and B) are time-shared between roll attitude error and bank rate command. During Ascent, it represents only roll attitude error. During Re-Entry, however, it represents roll attitude error when the computed range is less than the desired range, and a 20 degree per second bank rate command when the computed range equals or exceeds the desired range.

  (1) Roll attitude error (Ascent) to Attitude Display

  (2) Roll attitude error (Re-Entry) to Attitude Display and ACME

(3) Bank rate command (Re-Entry) to Attitude Display and ACME

(c) Yaw attitude error (XCLYCRM) and return (XCLYCRMG) - Two identical sets of outputs (A and B) are time-shared between yaw attitude error (during Ascent) and cross range error (during Re-Entry).

(1) Yaw attitude error (Ascent) to Attitude Display

(2) Cross range error (Re-Entry) to Attitude Display

Titan Autopilot (Figure 8-36): During Ascent, the computer performs guidance computations in parallel with the Titan guidance and control system. If a malfunction occurs in the Titan system, the pilot can switch control to the Inertial Guidance System. For a description of the program requirements and operation associated with the Titan Autopilot interface, refer to the Attitude Display and ACME interface description.

The computer outputs to the Titan Autopilot are summarized as follows:

(a) Pitch error (XCLPDC) -
(b) Roll error (XCLRDC) -
(c) Yaw error (XCLYDC) -
(d) Common return (XCLDCG) -

These signals are provided during backup ascent guidance.

(e) Autopilot scale factor (XCDAPSF) - This signal changes the autopilot dynamics after the point of maximum dynamic pressure is reached.

(f) Second stage engine cutoff (XCDSSCF) - This signal is generated when velocity to be gained equals zero.

Pilots' Control and Display Panel (PCDP) (Figure 8-37)

The following CLD instruction programming is associated with the PCDP interface:

Signal	Address X	Address Y
Computer mode 1	1	1
Computer mode 2	0	1
Computer mode 3	3	1
Start computation	1	2
Abort transfer	7	1
Fade-in discrete	6	1

The following PRO instruction programming is associated with the PCDP interface:

Signal	Address X	Address Y
Computer malfunction	4	3
Computer running	5	0
Reset start computation	2	6

The computer inputs from the PCDP are summarized as follows:

(a) Computer on (XHONP) and computer off (XHOFF) - These signals from the Computer On-Off switch control computer power.

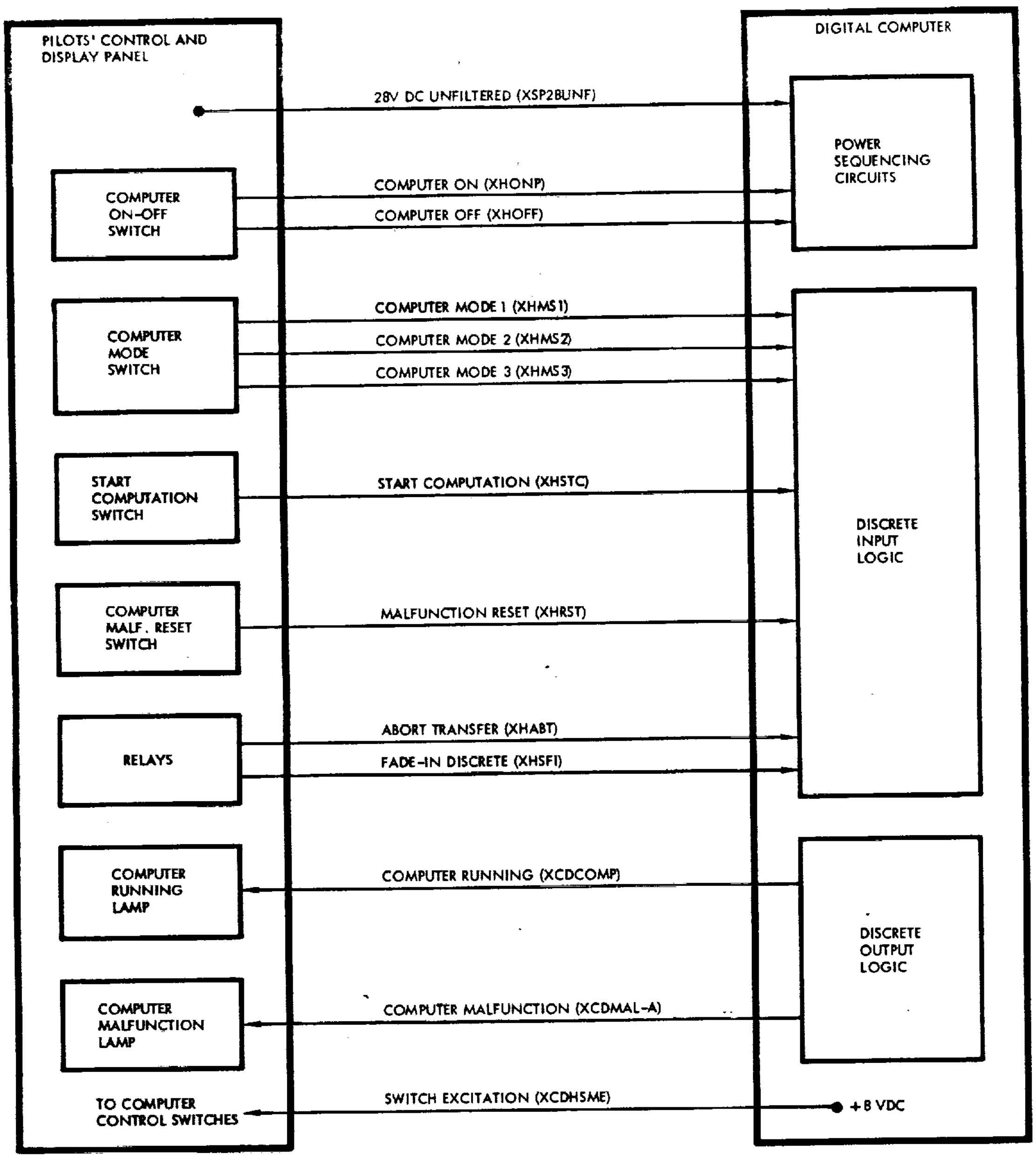

Figure 8-37 Computer-PCDP Interface

(b) Computer mode - The computer receives three binary coded discrete signals from the Computer Mode switch, to define the following operational modes:

Mode	Computer Mode 1 (XHMS1)	Computer Mode 2 (XHMS2)	Computer Mode 3 (XHMS3)
Pre-Launch	"0"	"0"	"1"
Ascent	"0"	"1"	"0"
Re-Entry	"1"	"0"	"1"

(c) Start computation (XHSTC) - This signal from the Start Computation push-button switch initiates re-entry calculations.

(d) Malfunction reset (XHRST) - This signal from the Computer Malfunction Reset switch resets the computer malfunction latch. The pilot uses the switch to test for a transient failure.

(e) Abort transfer (XHABT) - The signal automatically switches the computer from the Ascent mode to the Re-Entry mode.

(f) Fade-in discrete (XHSFI) - This signal from a relay is supplied to the accumulator via the discrete input logic.

(g) 28 VDC Unfiltered (XSP28UNF)

The computer outputs to the PCDP are summarized as follows:

(a) Computer running (XCDCOMP) - This program-controlled signal lights the Computer Running lamp which is used as follows:

(1) Pre-Launch: The Computer Running lamp remains off during this mode, except during mission simulation when its operation is governed by the mode being simulated.

(2) Ascent: The Computer Running lamp turns on following Inertial Platform release. The lamp remains on for the duration of the mode, and then turns off.

NOTE

For a description of lamp operation during the Catch-up and Rendezvous modes, refer to Vol. II of this document.

(3) Re-Entry: The Computer Running lamp lights when the Start Computation push-button switch is depressed or when time to start re-entry calculations is equal to zero. The lamp remains on for the duration of the mode, and then turns off.

(b) Computer malfunction (XCDMAL-A) - This signal turns on the Computer Malfunction lamp. Either the computer diagnostic program, a built-in timing check, or an AGE command actuates the signal.

(c) Switch excitation (XCDHSME) - This DC excitation is supplied to the Computer Mode switch, the Start Computation switch, and the Malfunction Reset switch.

Incremental Velocity Indicator (IVI) (Figure 8-38)

The IVI contains three incremental velocity counters that display velocity increments along the spacecraft (body) axes.

Power is applied to the IVI whenever the computer is turned on. During the first 30-second period (or less) following the application of power, the IVI automatically references its counters to zero. After this period, the IVI is capable of recognizing computer signals.

The IVI counters can be set manually by means of control knobs on the front of the unit, or they can be set automatically by the computer. After the counters are initially set, they are driven by incremental velocity pulses from the computer. These pulses are used to update the indications displayed by the counter. The computer can set the individual counters to zero by providing a 20 usec pulse on each of three set zero lines. A feed-back signal, denoting zero counter position, permits the computer to test for the proper counter reference prior to the insertion and display of a computed velocity increment.

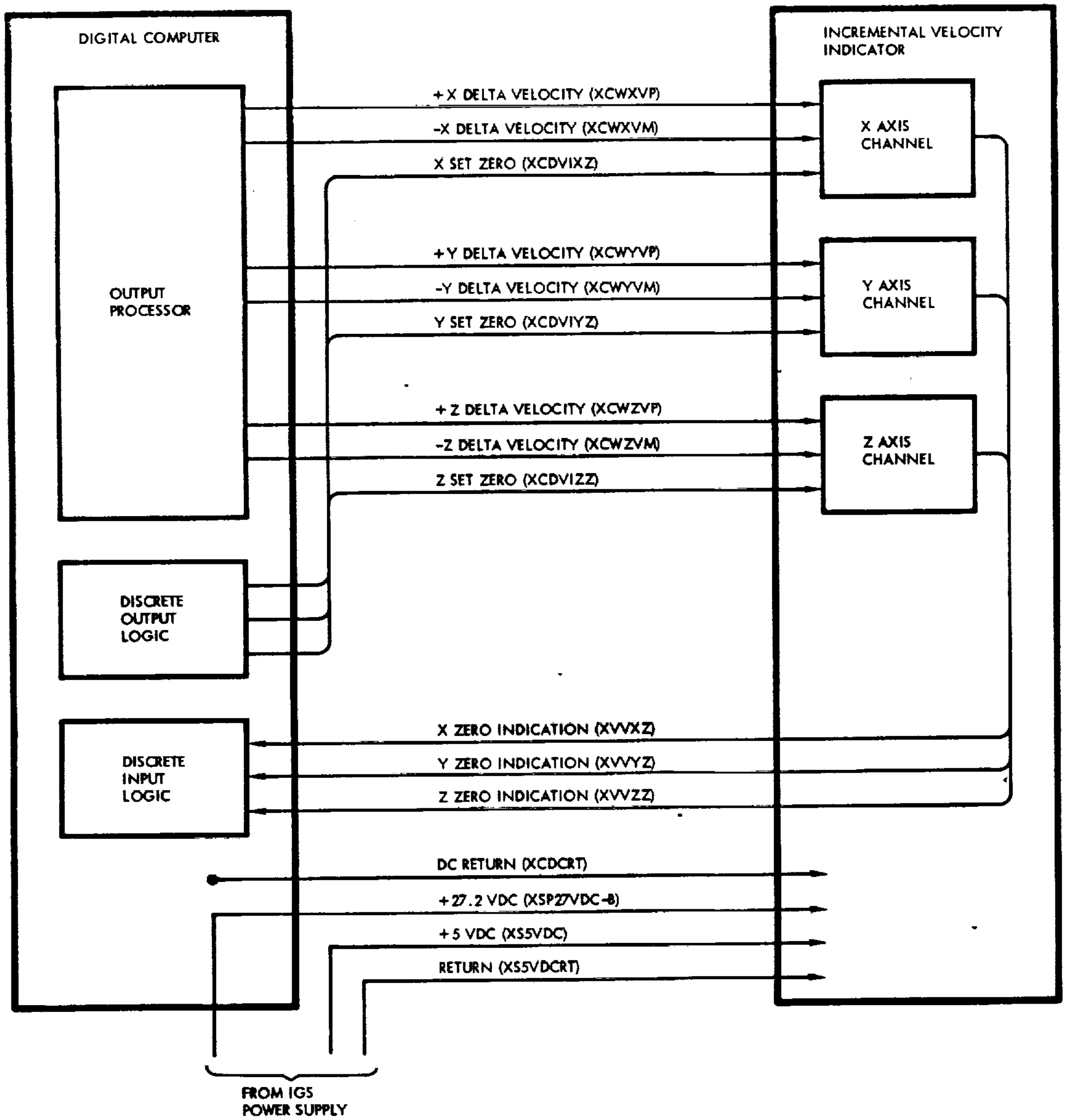

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Figure 8-38 Computer-IVI Interface

The following CLD instruction programming is associated with the IVI interface:

Signal	Address X	Address Y
X zero indication	1	3
Y zero indication	5	2
Z zero indication	6	2
Velocity error count not zero	2	2

The following PRO instruction programming is associated with the IVI interface:

Signal	Address X	Address Y
Select X counter	2	1
Select Y counter	3	1
Drive counters to zero	1	1
Write output processor	5	3

The computer supplies three signals to the IVI, one for each counter, that are used to position the counters to zero. To generate these signals, the program sets DO11 minus and sets DO12 and DO13 as follows:

Signal	DO12	DO13
X set zero	Minus	Plus
Y set zero	Plus	Minus
Z set zero	Minus	Minus

The IVI provides three feed-back signals to the computer (DI31, DI25, and DI26) to indicate that the counters are zeroed. The program tests the individual counters for zero position before attempting to drive them to zero.

The output processor provides a timed output to the IVI that represents velocity increments along the spacecraft axes. One output channel (phase 2) on the delay line is time-shared among the X, Y, and Z counters. Incremental velocities (in two's-complement form) are written on the delay line during phase 2 from accumulator bit positions S, and 1 through 12. Discrete outputs DO12 and DO13, which are set no more than 1 ms before the PRO35 operation, select the proper velocity signal as follows:

Signal	DO12	DO13
X velocity	Minus	Plus
Y velocity	Plus	Minus
Z velocity	Minus	Minus

Once data is written on the delay line, the output of the delay line is sensed for data during bit times BT1 through BT12. Any bit sensed during this time indicates the presence of data which is then gated into a buffer along with the sign bit (BT13) during phase 2. This buffer is sampled approximately every 21.5 ms and a pulse is generated if the buffer is set either plus or minus. During this same time, an update cycle is initiated and a count of one is either added to or subtracted from the delay line data to decrease the magnitude by a count of one. If the buffer is set to zero during the update cycle, the data on the delay line is recirculated without affecting its magnitude.

The zero output of the buffer is addressed as DI22. When this discrete input is off, velocity data has been counted down to zero and the next velocity can be processed.

The computer inputs from the IVI are summarized as follows:

(a) X zero indication (XVVXZ) - The down level signifies that the X channel of the IVI is at the zero position.

(b) Y zero indication (XVVYZ) - The down level signifies that the Y channel of the IVI is at the zero position.

(c) Z zero indication (XVVZZ) - The down level signifies that the Z channel of the IVI is at the zero position.

The computer outputs to the IVI are summarized as follows:

(a) +X delta velocity (XCWXVP) - The up level denotes that the X channel should change by one foot per second in the fore direction.

(b) -X delta velocity (XCWXVM) - The up level denotes that the X channel should change by one foot per second in the aft direction.

(c) X set zero (XCDVIXZ) - The up level drives the X channel to the zero position.

(d) +Y delta velocity (XCWYVP) - The up level denotes that the Y channel should change by one foot per second in the right direction.

(e) -Y delta velocity (XCWYVM) - The up level denotes that the Y channel should change by one foot per second in the left direction.

(f) Y set zero (XCDVIYZ) - The up level drives the Y channel to the zero position.

(g) +Z delta velocity (XCWZVP) - The up level denotes that the Z channel should change by one foot per second in the down direction.

(h) -Z delta velocity (XCWZVM) - The up level denotes that the Z channel should change by one foot per second in the up direction.

(i) Z set zero (XCDVIZZ) - The up level drives the Z channel to the zero position.

Instrumentation System (IS) (Figure 8-39)

The computer is interfaced with the Multiplexer Encoder Unit (MEU) and the Signal Conditioning Equipment (SCE) of the IS. Continuous analog data is provided to the SCE and stored digital quantities are sent upon request to the MEU.

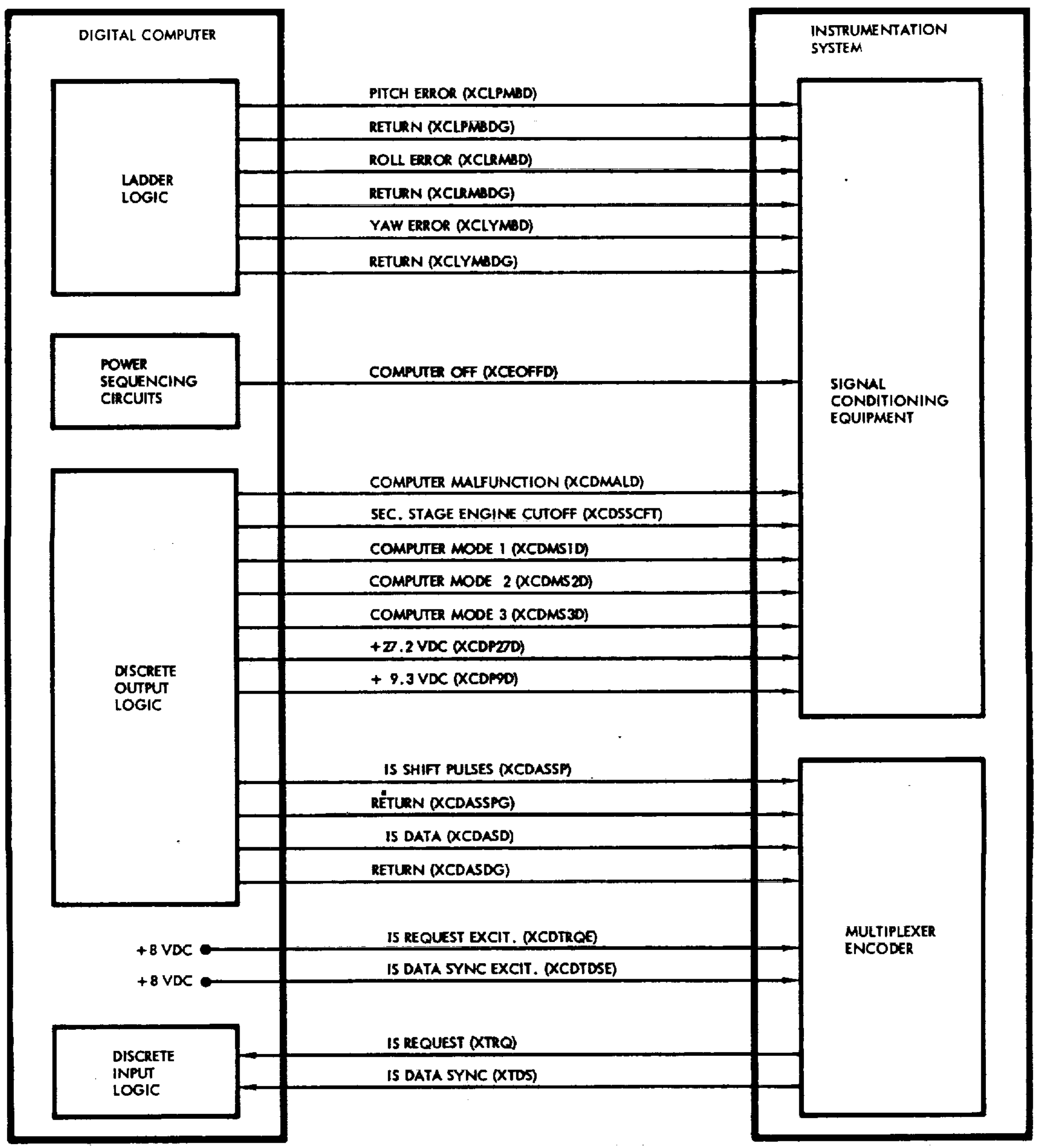


Figure 8-39 Computer-IS Interface

Certain computer data, as described below, is continually made available to the SCE. The SCE conditions this data for multiplexing and analog-to-digital conversion by the MEU.

(a) Computer modes - The mode signals transmitted to the computer are monitored to determine that the computer was in the correct mode for a particular operational mission phase.

(b) Computer input power - The 27.2 VDC and 9.3 VDC inputs supplied to the computer by the IGS Power Supply are monitored via the computer.

(c) Computer off - The output of the off position of the Computer On-Off switch is monitored via the computer. (S/C 3)

(d) Computer running - The computer running discrete output is monitored and recorded (S/C 4 and 7)

(e) Computer malfunction - The compúter malfunction discrete output is monitored and recorded.

(f) Attitude errors: The pitch, yaw, and roll AC analog attitude errors are monitored and recorded.

Twenty-one data word locations in the computer memory are allocated for the storage of IS data. Data stored in these locations is dependent upon the computer mode of operation.

The following CLD instruction programming is associated with the IS interface:

Signal	Address	
	X	Y
IS request	7	0
IS sync	2	1

The following PRO instruction programming is associated with the IS interface:

Signal	Address	
	X	Y
IS control gate	0	1

Every 50 ms or less, the computer program tests the IS request discrete input (DI07). If the discrete input is tested minus, the IS sync discrete input (DI12) is tested as follows:

(a) DI12 minus - The program stores current specified values, according to the computer mode, in an IS memory buffer of 21 locations. The contents of the first buffer location are placed in the accumulator so that the sign position of the data word corresponds to the sign position of the accumulator. Then a PRO10 instruction is given. This instruction causes the information contained in accumulator bit positions S, and 1 through 23 to be supplied to the IS. Twenty-four shift pulses are also supplied to the IS.

(b) DI12 plus - An IS program counter is incremented by one and the contents of the next sequential buffer location are placed in the accumulator and sent to the IS via PRO10 instructions. Subsequent IS requests advance the program counter until all 21 instrumentation quantities are transmitted.

The computer inputs from the IS are summarized as follows:

(a) IS request (XTRQ) - An up level on this line signifies that the IS requires a computer data word. The word is transferred from the computer within 75 ms of the request. Requests can occur at rates up to 10 times per second.

(b) IS data sync (XTDS) - An up level on this line signifies the beginning of the IS data transfer operation.

The computer outputs to the IS are summarized as follows:

(a) IS shift pulses (XCDASSP) and return (XCDASSPG) - This series of 24 pulses causes IS data to be transferred to the IS buffer.

(b) IS data (XCDASD) and return (XCDASDG) - These 24 bits of data are transferred in synchronism with the IS shift pulses.

(c) IS request excitation (XCDTRQE) - This +8 VDC signal is the excitation for the IS request signal.

(d) IS data sync excitation (XCDTDSE) - This +8 VDC signal is the excitation for the IS data sync signal.

(e) Monitored signals - The following signals are supplied to the IS for monitoring purposes:

(1) Pitch error (XCLPMBD) and return (XCLPMBDG)
(2) Roll error (XCLRMBD) and return (XCLRMBDG)
(3) Yaw error (XCLYMBD) and return (XCLYMBDG)
(4) Computer off (XCEOFFD)
(5) Computer malfunction (XCDMALD)
(6) Second stage engine cutoff (XCDSSCFT)
(7) Computer mode 1 (XCDMS1D)
(8) Computer mode 2 (XCDMS2D)
(9) Computer mode 3 (XCDMS3D)
(10) +27.2 VDC (XCDP27D)
(11) +9.3 VDC (XCDP9D)

Aerospace Ground Equipment (AGE) (Figure 8-40)

The AGE determines spacecraft-installed computer status by being able to read and display the contents of any memory location, initiate and terminate marginal tests of the memory timing, and command the computer to condition the computer malfunction circuit. These tests are accomplished by a hard-wired computer/AGE data link.

In conjunction with a voice link to the spacecraft, the AGE can control the various computer modes of operation to determine the status of the computer and its interfaces. To aid in localizing failures, the AGE monitors the following computer signals:

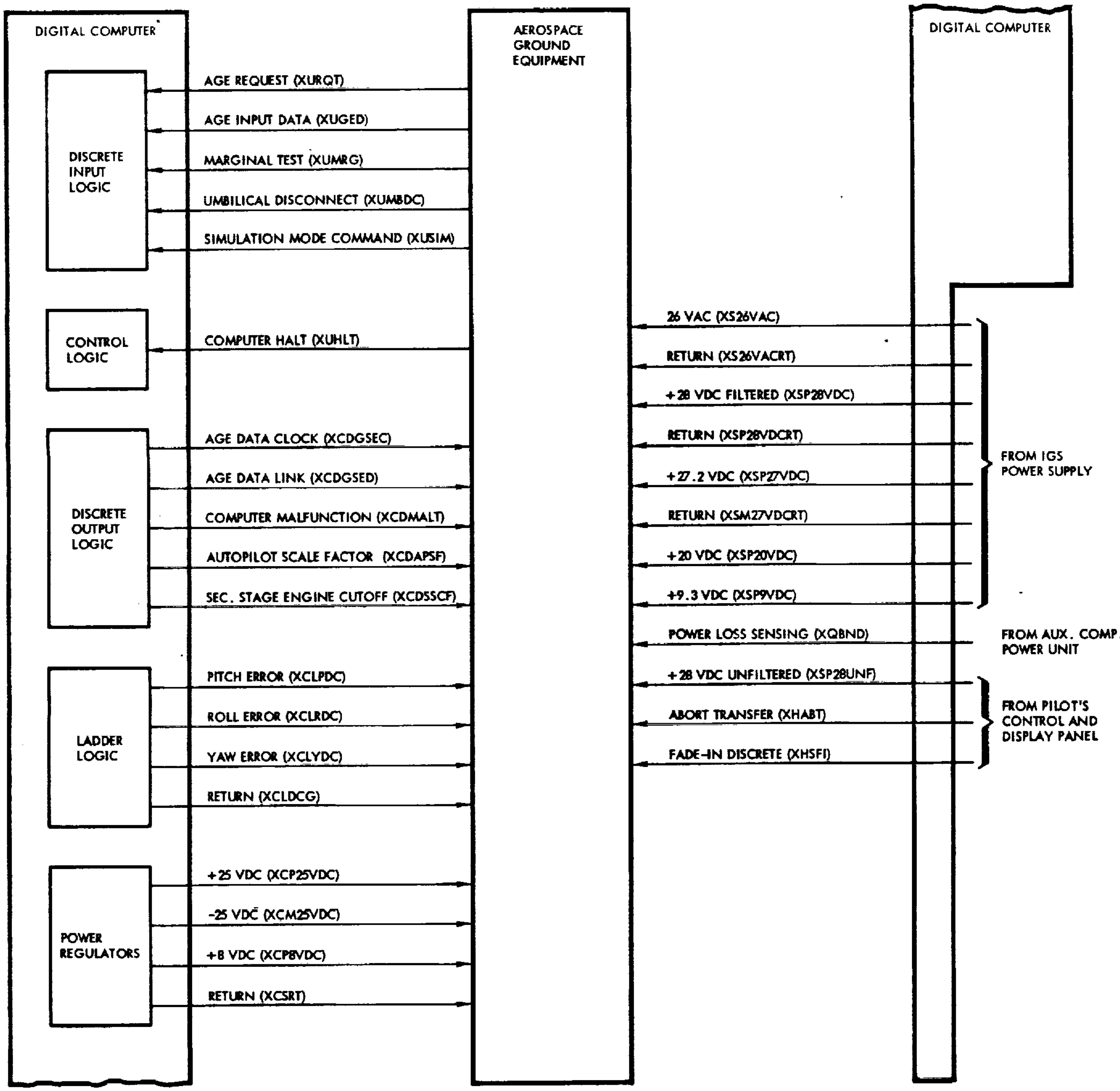


Figure 8-40 Computer-AGE Interface

FM2-8-42

(a) All input and output voltages

(b) Second stage engine cutoff

(c) Autopilot scale factor

(d) Roll error command

(e) Yaw error command (to Titan Autopilot)

(f) Pitch error command

(g) Computer malfunction

In addition, the AGE provides two hard-wired inputs to the computer to reset the malfunction circuit and half the computer and to force a marginal check of the memory timing. Early and late strobing of the memory is effected using the computer/AGE data link.

The following CLD instruction programming is associated with the AGE interface:

Signal	Address X	Address Y
AGE request	2	3
AGE input data	7	2
Simulation mode command	4	2
Umbilical disconnect	6	3

The following PRO instruction programming is associated with the AGE interface:

Signal	Address X	Address Y
AGE data link	2	2
AGE data clock	3	2
Computer malfunction	4	3
Memory strobe	0	6
Autopilot scale factor	1	6
Second stage engine cutoff	4	6

The AGE program commences when the AGE request (DI32) is tested minus. To receive the 18 bit AGE data word, the program repeats the following sequence of operations 18 times:

(a) Turn on AGE data clock (DO23)

(b) Wait 2.5 ms

(c) Reset AGE data clock (DO23)

(d) Wait 1.5 ms

(e) Read AGE input data (DI27)

(f) Wait 1.5 ms

The above sequence causes the 18-bit AGE word to be shifted out of the AGE register and into the computer. The first 4 bits of the AGE word are mode bits, and the remaining 14 bits are data. The coding of the 4 mode bits is as follows:

Mode Bits				Mode
4	3	2	1	
0	0	0	0	None
0	0	0	1	Read any word
0	0	1	0	Set marginal early
0	0	1	1	Set computer malfunction on
0	1	0	0	Set marginal late
0	1	0	1	Set pitch ladder output
0	1	1	0	Set yaw ladder output
0	1	1	1	Set roll ladder output
1	0	0	0	Set all ladder outputs

In the read any word mode, the 14 data bits of the AGE word are as follows:

18	17	16	15	14	13	12	11	10	9	8	7	6	5
S5	S4	S3	S2	S1	A9	A8	A7	A6	A5	A4	A3	A2	A1

where A1 through A8 define the address of the requested data, A9 sets up AGE internal clock pulse timing, S1 through S4 define the sector of the requested data, and S5 defines the syllable(s) of the requested data. The computer determines the requested data and sends it to the AGE. If the requested data is located in syllables 0 and 1, it is sent to the AGE starting with the high-order bit of syllable 1 and finishing with the low-order bit of syllable 0. If the requested data is located in syllable 2, the first 13 bits sent to the AGE are "0's," and the last 13 bits are data from syllable 2 (high-order bit first). Requested data is sent to the AGE by executing the following sequence of operations 26 times. There is a delay of 4.5 ms between resetting clock 18 and setting clock 19.

(a) Set AGE data link (D022) from accumulator sign position

(b) Turn on AGE data clock (D023)

(c) Wait 2.5 ms

(d) Reset AGE data clock (D023)

(e) Wait 2 ms

(f) Reset AGE data link (D022)

(g) Wait 1 ms

In the set marginal early mode, the computer sets D060 on. This signal, in conjunction with the marginal test signal provided by the AGE, causes early strobing of the computer memory.

In the set computer malfunction on mode, the computer sets D034 on to check the malfunction indication.

In the set marginal late mode, the computer sets D060 off. This signal, in conjunction with the marginal test signal, causes late strobing of the computer memory.

In the set ladder outputs modes, the 14 data bits of the AGE word are as follows:

18	17	16	15	14	13	12	11	10	9	8	7	6	5
S	D6	D5	D4	D3	D2	D1	0	0	0	0	0	0	0

where D1 through D6 are data bits and S is the sign bit. The data and sign bits are used to control the ladder output indicated by the 4 associated mode bits. The number is in two's-complement form where D1 is the low-order data bit.

The computer inputs from the AGE are summarized as follows:

(a) AGE request (XURQT) - An up level signifies that the AGE is ready to transfer a message to the computer.

(b) AGE input data (XUGED) - An up level denotes a binary "1" being transferred from the AGE to the computer.

(c) Marginal test (XUMRG) - An up level, in conjunction with the proper AGE message, causes the computer memory timing to be marginally tested.

(d) Umbilical disconnect (XUMBDC) - An open circuit on this line signifies that the Inertial Platform has been released (or that the torquing signals have been removed). The Inertial Platform then enters the inertial mode of operation and the computer begins to perform the navigation guidance portion of its Ascent routine.

(e) Simulation mode command (XUSIM) - This command causes the computer to operate in a simulated mode as determined by the Computer Mode switch.

(f) Computer halt (XUHLT) - An up level resets the computer malfunction circuit and sets the computer halt circuit.

The computer outputs to the AGE are summarized as follows:

(a) AGE data clock (XCDGSEC) - This line reads out the AGE register and synchronizes the AGE with the AGE data link.

(b) AGE data link (XCDGSED) - An up level denotes a binary "1" being transferred from the computer to the AGE.

(c) Computer malfunction (XCDMALT) - An up level indicates that the computer malfunction latch is set. The latch can be set by the computer diagnostic program, a timing error, program looping, or an AGE command.

(d) Monitored signals - The following signals and voltages are supplied to the AGE for monitoring or recording purposes:

(1) Autopilot scale factor (XCDAPSF)

(2) Second stage engine cutoff (XCDSSCF)

(3) Pitch error (XCLPDC)
(4) Yaw error (XCLYDC)
(5) Roll error (XCLRDC)
and common return (XCLDCG)

(6) +25 VDC (XCP25VDC)
(7) -25 VDC (XCM25VDC)
(8) +8 VDC (XCP8VDC)
and common return (XCSRT)

(9) 26 VAC (XS26VAC) and return (XS26VACRT)

(10) +28 VDC filtered (XSP28VDC) and return (XSP28VDCRT)

(11) +28 VDC unfiltered (XSP28UNF)

(12) +27.2 VDC (XSP27VDC)

(13) -27.2 VDC return (XSM27VDCRT)

(14) +20 VDC (XSP20VDC)

(15) +9.3 VDC (XSP9VDC)

(16) Power loss sensing (XQBND)

(17) Abort transfer (XHABT)

(18) Fade-in discrete (XHSFI)

MANUAL DATA INSERTION UNIT

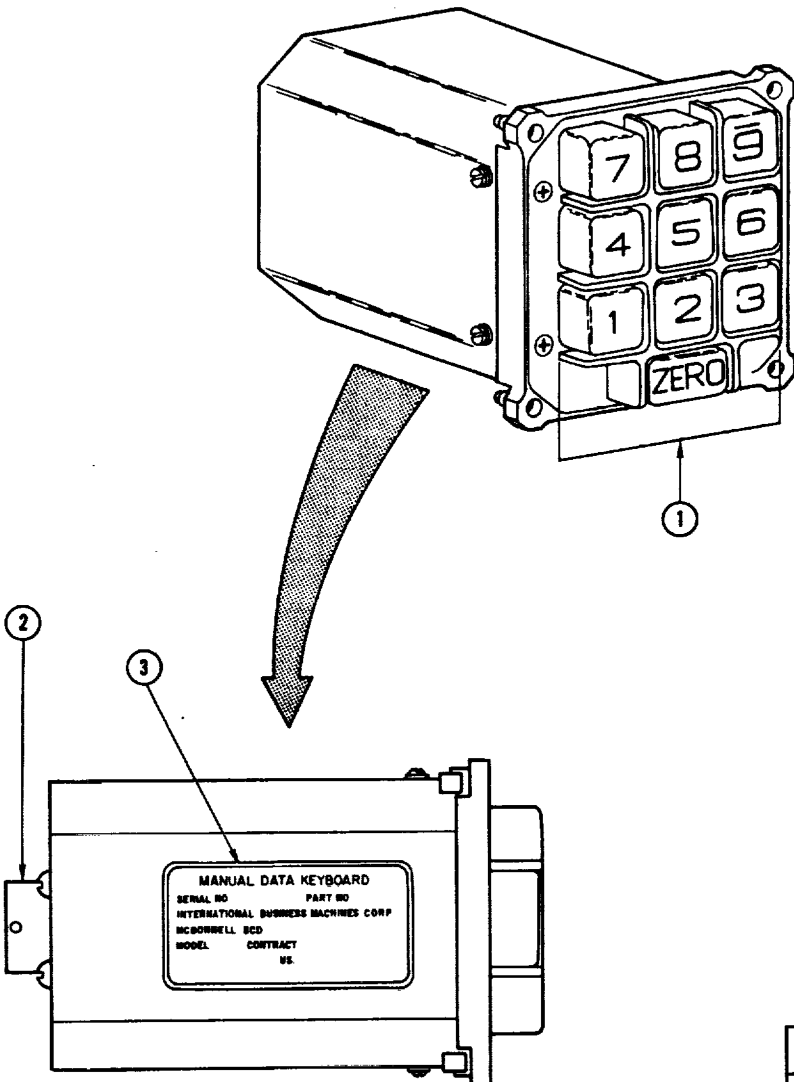
SYSTEM DESCRIPTION

Purpose

The Manual Data Insertion Unit (MDIU) physically consists of the Manual Data Keyboard (MDK) (Figure 8-41) and the Manual Data Readout (MDR) (Figure 8-42) respectively. The MDIU enables the pilot to insert data into, and read data from, the computer memory.

Performance

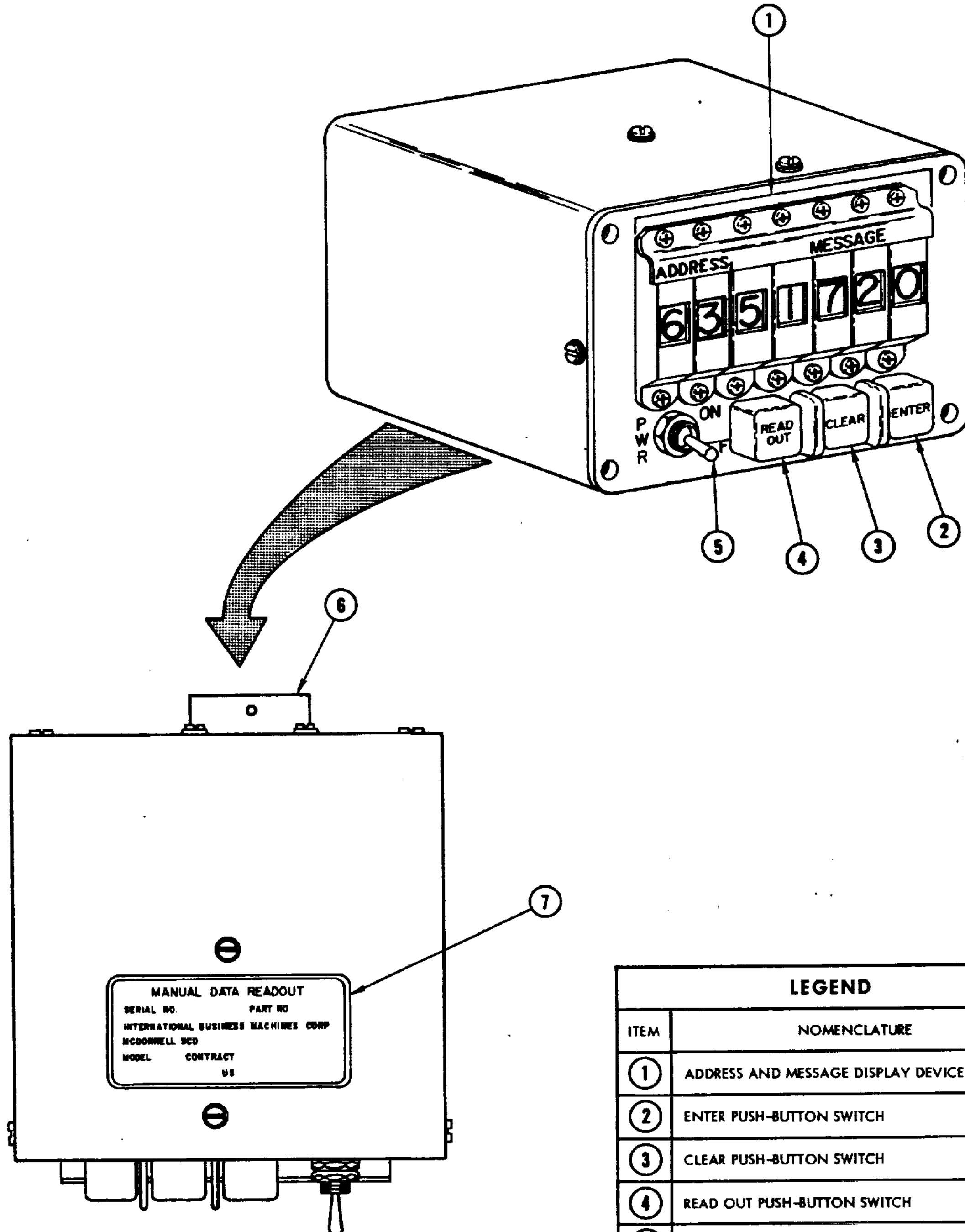
Data Insertion: Before data is set up for insertion into the computer, all existing data is cleared from the MDIU by pressing the CLEAR push-button switch on the MDR. Then the Data Insert push-button switches on the MDK are used to set up a 7-digit decimal number. The first two digits from the left specify the address of the computer memory location in which the data is to be stored, and the last five digits specify the data itself. As the data is set up, it



LEGEND	
ITEM	NOMENCLATURE
1	DATA INSERT PUSH-BUTTON SWITCHES
2	CONNECTOR J1
3	IDENTIFICATION PLATE

FM2-8-43

Figure 8-41 Manual Data Keyboard



LEGEND	
ITEM	NOMENCLATURE
1	ADDRESS AND MESSAGE DISPLAY DEVICES
2	ENTER PUSH-BUTTON SWITCH
3	CLEAR PUSH-BUTTON SWITCH
4	READ OUT PUSH-BUTTON SWITCH
5	PWR (POWER) TOGGLE SWITCH
6	CONNECTOR J1
7	IDENTIFICATION PLATE

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Figure 8-42 Manual Data Readout

is automatically supplied to the computer accumulator. A digit-by-digit verification of the address and data is made by means of the ADDRESS and MESSAGE display devices on the MDR. After verification, the ENTER push-button switch on the MDR is pressed to store the data in the selected memory location.

Data Readout

Before data is read from the computer, all existing data is cleared from the MDIU by pressing the CLEAR push-buttton switch. Then the Data Insert push-button switches are used to set up a 2-digit decimal number. The two digits specify the address of the computer memory location from which data is to be read. A digit-by-digit verification of the address is made by means of the ADDRESS display devices. After verification, the READ OUT push-button switch on the MDR is pressed and the data is read from the selected memory location and displayed by the MESSAGE display devices.

MDK Physical Description

The MDK is 3.38 inches high, 3.38 inches wide, and 5.51 inches deep. It weighs 1.36 pounds. External views of the MDK are shown on Figure 8-41. The major external characteristics are summarized in the accompanying legend.

MDR Physical Description

The MDR is 3.26 inches high, 5.01 inches wide, and 6.41 inches deep. It weighs 3.15 pounds. External views of the MDR are shown on Figure 8-42. The major external characteristics are summarized in the accompanying legend.

Controls and Indicators

The controls and indicators located on the MDK and MDR are illustrated on Figure 8-43. The accompanying legend identifies the controls and indicators, and describes their purposes.

SYSTEM OPERATION

Power

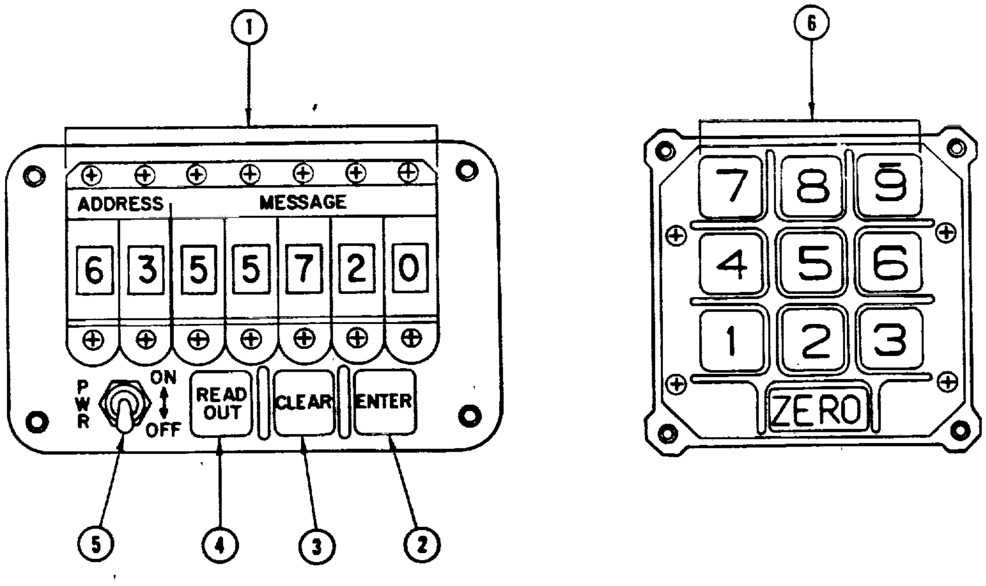
The MDIU receives all of the power required for its operation from the computer. This power consists of the following regulated DC voltages:

(a) +25 VDC and common return
(b) -25 VDC and common return
(c) +8 VDC and return

This power is available at the MDIU whenever the computer is turned on. However, it is not actually applied to the MDIU circuits until the POWER switch on the MDR is turned on. When power is turned on at the MDR, the regulated DC voltages are filtered by a capacitor network and supplied to the MDK and MDR circuits.

MDK Data Flow (Figure 8-44)

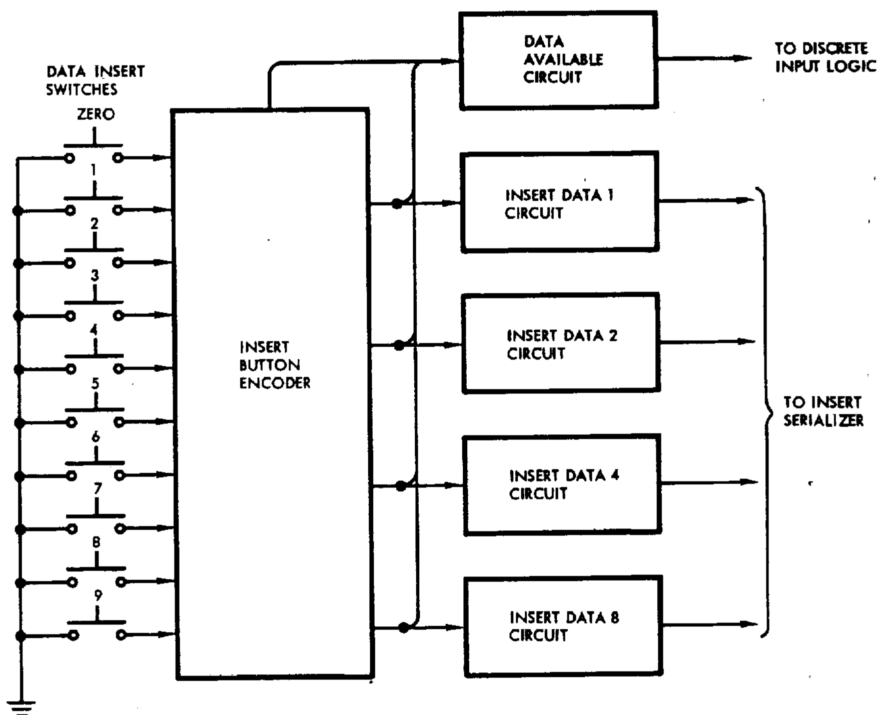
The MDK has ten Data Insert push-button switches. These switches are used to select the address of a computer memory location in which data is to be stored or from which data is to be read. For storing data, the push-button switches are also used to set up the actual data to be stored. Since the push-button switches are numbered decimally, the insert button encoder is used to convert their outputs to binary coded decimal values that can be used by the computer. These values, called the insert data signals, are supplied to the



LEGEND		
ITEM	NOMENCLATURE	PURPOSE
1	ADDRESS AND MESSAGE DISPLAY DEVICES	DISPLAY ADDRESS AND MESSAGE SENT TO COMPUTER DURING ENTER OPERATION; DISPLAY ADDRESS SENT TO, AND MESSAGE RECEIVED FROM, COMPUTER DURING READOUT OPERATION.
2	ENTER PUSH-BUTTON SWITCH	PROVIDES MEANS FOR CAUSING MESSAGE SENT TO COMPUTER DURING ENTER OPERATION TO BE STORED IN MEMORY.
3	CLEAR PUSH-BUTTON SWITCH	PROVIDES MEANS FOR CAUSING ADDRESS AND MESSAGE SET UP BY MDK TO BE CLEARED OR CANCELED.
4	READ OUT PUSH-BUTTON SWITCH	PROVIDES MEANS FOR CAUSING MESSAGE TO BE READ OUT OF COMPUTER AND DISPLAYED BY MESSAGE DISPLAY DEVICES.
5	PWR (POWER) TOGGLE SWITCH	PROVIDES MEANS FOR CONTROLLING APPLICATION OF POWER TO MDK AND MDR.
6	DATA INSERT PUSH-BUTTON SWITCHES	PROVIDE MEANS FOR CAUSING ADDRESS AND MESSAGE TO BE SENT TO COMPUTER AND TO BE DISPLAYED BY ADDRESS AND MESSAGE DISPLAY DEVICES.

FM2-8-45

Figure 8-43 Manual Data Insertion Unit Front Panels



FM2-8-46

Figure 8-44 Manual Data Keyboard Data Flow insert serializer in the computer. The insert button encoder also develops the data available signal which is supplied to the discrete input logic of the computer.

MDR Data Flow (Figure 8-45

The MDR has seven digital display devices and three command push-button switches. The display devices are used to display the address set up by the Data Insert push-button switches on the MDK, and to display either the data set up by the Data Insert push-button switches or the data read from a computer memory location. The command push-button switches, called ENTER, READ OUT, and CLEAR, are used to determine whether data is entered into or read out of the computer, or whether the data that has been set up is to be cleared (or canceled). These push-button switches all supply inputs to the discrete input logic of the computer. Since the display devices provide a decimal display, the binary coded decimal values received from the computer must be decoded before they can be displayed. These values from the computer are supplied to three device select control circuits and three insert data circuits. Another signal from the computer is supplied to the display device drive control circuit. A combination of outputs from the device select control circuits is used in conjunction with the outputs of the display device drive control circuit to select a particular display device. This selection is accomplished by means of the device selector. A combination of outputs from the insert data circuits is used in conjunction with the outputs of the display device drive control circuit to select a particular number on the selected display device. This selection

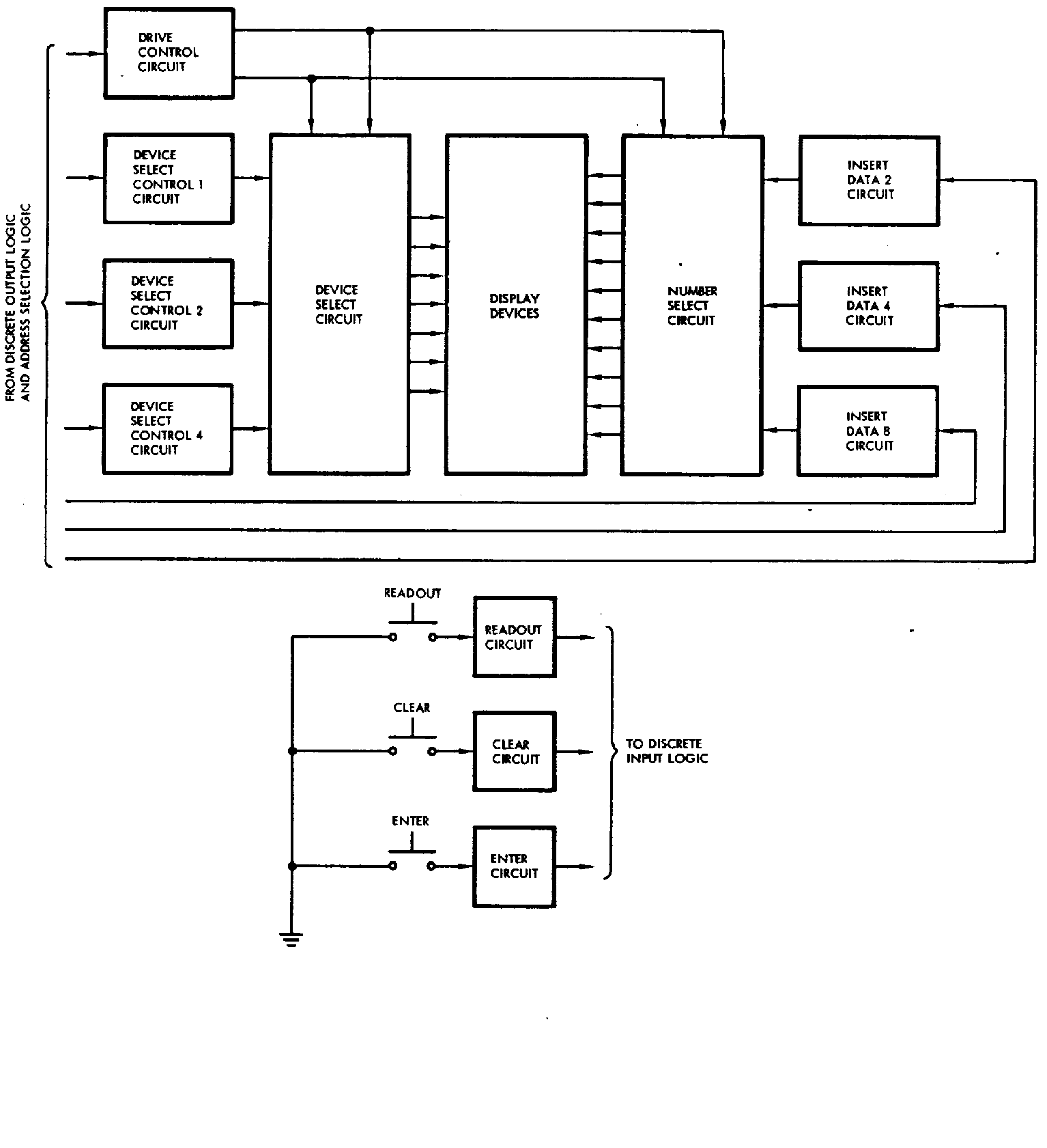


Figure 8-45 Manual Data Readout Data Flow

is accomplished by means of the number selector. Thus, through the combined operations of the device selector and the number selector, the binary coded decimal values received from the computer are decoded and an equivalent decimal display is presented on the display devices.

Manual Data Subroutine

The Manual Data subroutine, which determines when data is transferred between the MDIU and the computer, is described under the Operational Program heading in the DIGITAL COMPUTER SYSTEM OPERATION part of this section.

Interfaces

The MDIU interfaces, all of which are made with the computer, are described under the Interfaces heading in the DIGITAL COMPUTER SYSTEM OPERATION part of this section.

INCREMENTAL VELOCITY INDICATOR

SYSTEM DESCRIPTION

Purpose

The Incremental Velocity Indicator (IVI) (Figure 8-46) provides visual indications of incremental velocity for the longitudinal (forward-aft), lateral (left-right), and vertical (up-down) axes of the spacecraft. These indicated incremental velocities represent the amount and direction of additional velocity or thrust necessary to achieve correct orbit, and thus are added to the existing spacecraft velocities by means of the maneuver thrusters.

LEGEND	
ITEM	NOMENCLATURE
1	FWD (FORWARD) DIRECTION INDICATION LAMP
2	FORWARD-AFT DISPLAY DEVICE
3	L (LEFT) DIRECTION INDICATION LAMP
4	LEFT-RIGHT DISPLAY DEVICE
5	R (RIGHT) DIRECTION INDICATION LAMP
6	UP-DOWN DISPLAY DEVICE
7	UP DIRECTION INDICATION LAMP
8	DN (DOWN) DIRECTION INDICATION LAMP
9	DN-UP ROTARY SWITCH
10	L-R ROTARY SWITCH
11	AFT-FWD ROTARY SWITCH
12	AFT DIRECTION INDICATION LAMP
13	IDENTIFICATION PLATE
14	CONNECTOR J1

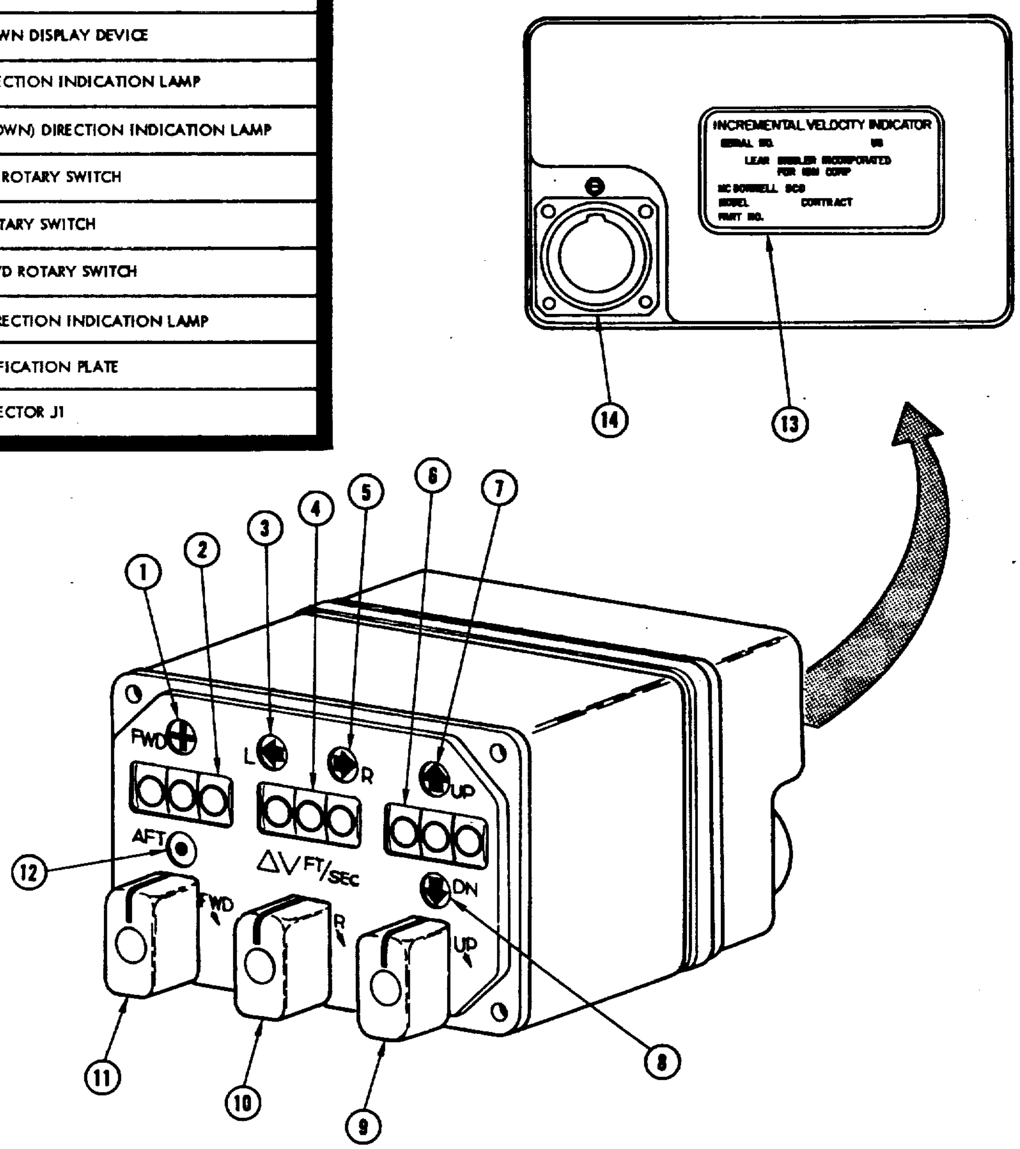


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Figure 8-46 Incremental Velocity Indicator

Performance

A three-digit decimal display device and two direction indication lamps are used to display incremental velocity for each of the three spacecraft axes. Both the lamps and the display devices can be set up either manually by rotary switches on the IVI or automatically by inputs from the computer. Then, as the maneuver thrusters correct the spacecraft velocities, pulses are received from the computer which drive the display devices toward zero. If a display device is driven beyond zero, indicating an overcorrection of the spacecraft velocity for the respective axis, the opposite direction indication lamp lights and the display device indication increases in magnitude to show a velocity error in the opposite direction.

Physical Description

The IVI is 3.25 inches high, 5.05 inches wide, and 5.89 inches deep. It weighs 3.25 pounds. External views of the IVI are shown on Figure 8-46. The major external characteristics are summarized in the accompanying legend.

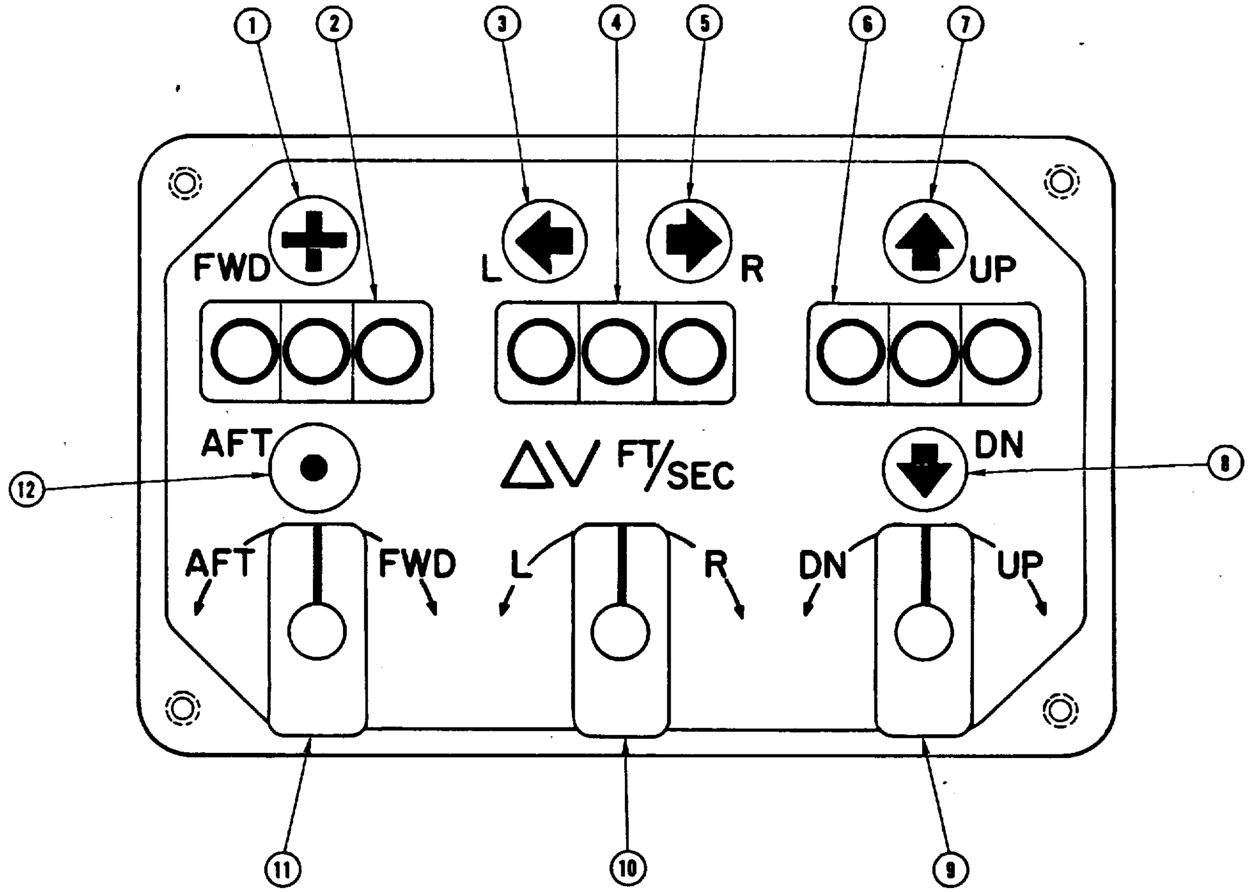
Controls and Indicators

The controls and indicators located on the IVI are illustrated on Figure 8-47. The accompanying legend identifies the controls and indicators, and describes their purposes.

SYSTEM OPERATION

Power

The power required for operation of the IVI is supplied by the IGS Power Supply whenever the computer is turned on. The power inputs are as follows:



LEGEND		
ITEM	NOMENCLATURE	PURPOSE
1	FWD (FORWARD) DIRECTION INDICATION LAMP	INDICATES THAT PLUS X AXIS VELOCITY IS INSUFFICIENT.
2	FORWARD-AFT DISPLAY DEVICE	INDICATES AMOUNT OF INSUFFICIENT VELOCITY FOR PLUS OR MINUS X AXIS.
3	L (LEFT) DIRECTION INDICATION LAMP	INDICATES THAT MINUS Y AXIS VELOCITY IS INSUFFICIENT.
4	LEFT-RIGHT DISPLAY DEVICE	INDICATES AMOUNT OF INSUFFICIENT VELOCITY FOR PLUS OR MINUS Y AXIS.
5	R (RIGHT) DIRECTION INDICATION LAMP	INDICATES THAT PLUS Y AXIS VELOCITY IS INSUFFICIENT.
6	UP-DOWN DISPLAY DEVICE	INDICATES AMOUNT OF INSUFFICIENT VELOCITY FOR PLUS OR MINUS Z AXIS.
7	UP DIRECTION INDICATION LAMP	INDICATES THAT MINUS Z AXIS VELOCITY IS INSUFFICIENT.
8	DN (DOWN) DIRECTION INDICATION LAMP	INDICATES THAT PLUS Z AXIS VELOCITY IS INSUFFICIENT.
9	DN-UP ROTARY SWITCH	PROVIDES MEANS FOR MANUALLY SETTING UP Z AXIS VELOCITY ERROR ON UP-DOWN DISPLAY DEVICE.
10	L-R ROTARY SWITCH	PROVIDES MEANS FOR MANUALLY SETTING UP Y AXIS VELOCITY ERROR ON LEFT-RIGHT DISPLAY DEVICE.
11	AFT-FWD ROTARY SWITCH	PROVIDES MEANS FOR MANUALLY SETTING UP X AXIS VELOCITY ERROR ON FORWARD-AFT DISPLAY DEVICE.
12	AFT DIRECTION INDICATION LAMP	INDICATES THAT MINUS X AXIS VELOCITY IS INSUFFICIENT.

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Figure 8-47 Incremental Velocity Indicator Front Panel

(a) +27.2 VDC and return

(b) +5 VDC and return

During the first 30 seconds (or less) following the application of power, the incremental velocity counters on the IVI are automatically driven to zero. Thereafter, the IVI is capable of normal operation.

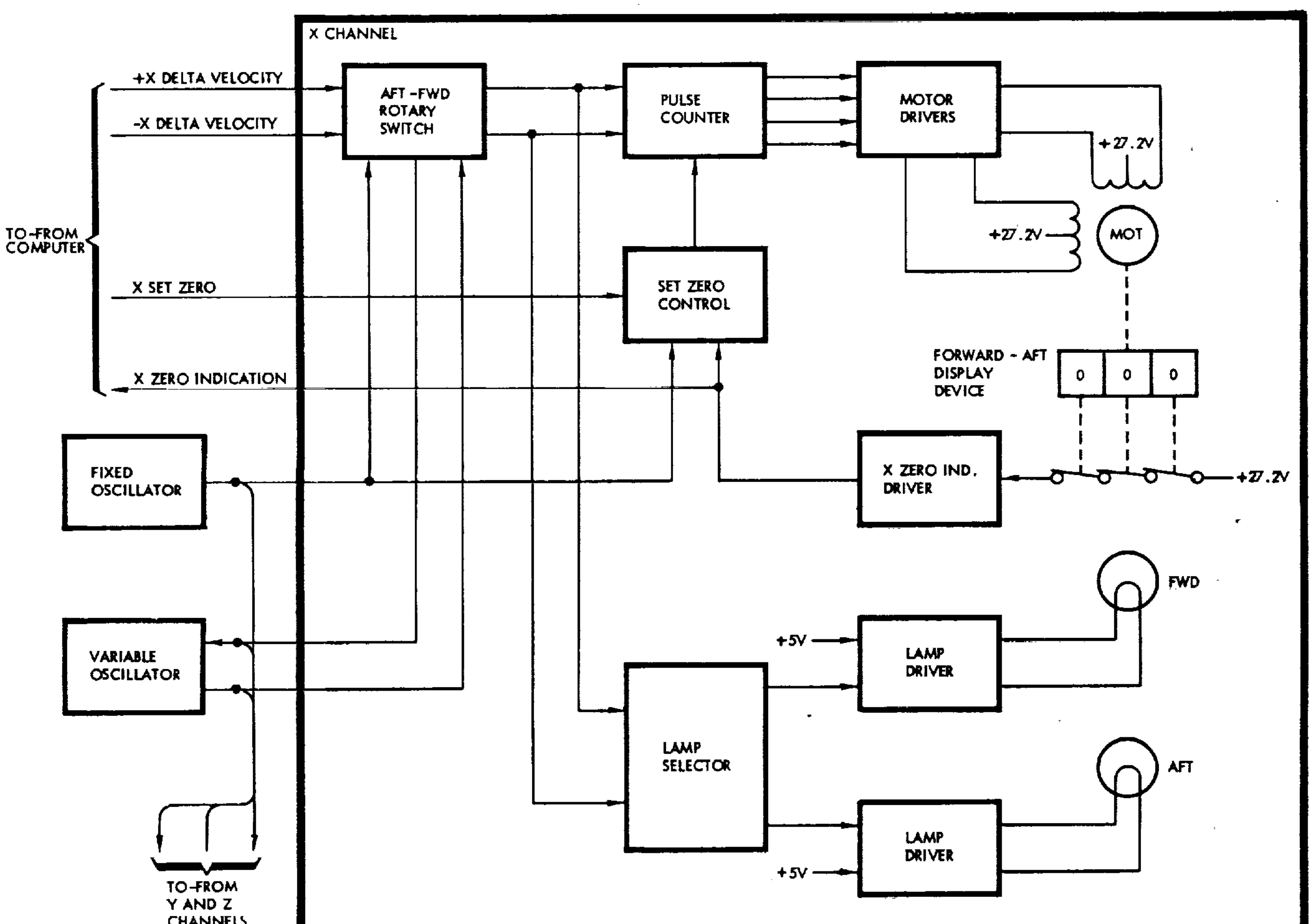
Basic Operation

The IVI includes three identical channels, each of which accepts velocity error pulses for one of the spacecraft axes and processes them for use by a decimal display device and its two associated direction indication lamps. The velocity error pulses are either received from the computer or generated within the IVI, as determined by the position of the rotary switch associated with each channel. With the spring-loaded switches in their neutral center positions, the IVI processes only the pulses received from the computer. However, rotation of the switches in either direction removes the pulses received from the computer and replaces them with pulses generated by an internal variable oscillator. These pulses are generated at a rate of one pulse per second for every 13.5 degrees of rotation until the rate reaches 10 pulses per second. Rotation of the switches beyond the 10 pulse per second position removes the pulses generated by the variable oscillator and replaces them with pulses generated by an internal fixed oscillator. These pulses are generated at a rate of 50 pulses per second. Rotation of the switches beyond the 50 pulse per second position is limited by mechanical stops.

The first pulse received on any channel, from either the computer or one of the oscillators, causes the appropriate display device to display a count of one. Simultaneously, this same pulse causes one of the two associated direction indication lamps to light. If the pulse was received on a positive input line, a forward, right, or down direction is indicated, depending on which channel (X, Y, or Z) received the pulse; and if the pulse was received on a negative input line, an aft, left, or up direction is indicated, depending on which channel received the pulse. Each additional pulse either increases or decreases the count depending on the relationship between the sign of the existing value on the counters and the sign of the added pulse as determined by the line on which it is received. A pulse having the same sign as the existing error increases the count; conversely, a pulse having the opposite sign of the existing error decreases the count. A series of pulses having the opposite sign indicates a corrective thrusting and eventually reduces the indicated error to zero. An overcorrection, causing still more pulses, causes the count to increase again but with the opposite direction indication lamp lit.

Zero Indication

As shown on Figure 8-48, three series-connected switches are operated by the Forward-Aft display device. (The same thing is true for the Y and Z channels; however, since the three channels are identical, only the X channel is shown.) When the display device indicates 000, all three switches are closed. A -27.2 VDC signal is then applied to the X zero indication driver which develops the X zero indication signal that is supplied to the computer. This signal indicates that the respective counter is at zero.



NOTE

Y AND Z CHANNELS ARE SAME AS X CHANNEL, EXCEPT Y CHANNEL CONTROLS AND INDICATORS ARE LEFT-RIGHT AND Z CHANNEL CONTROLS AND INDICATORS ARE UP-DOWN.

Figure 8-48 Incremental Velocity Indicator Data Flow

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Pulse Count

Velocity error pulses are applied to the lamp selector and the pulse counter via the AFT-FWD rotary switch. If the switch is in the center position, these pulses are received from the computer on the +X delta velocity line and the -X delta velocity line. If the switch is not in the center position, the pulses are received from either the fixed oscillator or the variable oscillator. As previously explained, the oscillator that is used depends on the exact position of the switch. Regardless of the source of the pulses, the lamp selector and the pulse counter operate the same. The lamp selector determines, by means of the sign of the error, which lamp should be lit. Power is then supplied to the selected lamp via the associated lamp driver. Meanwhile, the same pulses are being processed by the pulse counter and supplied to the motor drivers. The pulse counter and the motor drivers operate in a manner that causes the motor to be driven 90 degrees for each pulse that is counted. The direction in which the motor is driven is determined by the relationship between the sign of the existing velocity error count and the sign of the added velocity error pulse. The motor drives the display device so that it changes by a count of one for each 90 degrees of motor rotation. Thus the display device maintains an up-to-date count of the size of the velocity error for the associated axis (in this case, the X axis), and the direction indication lamps maintain an up-to-date indication of the direction of the error.

Zero Command

The IVI counters can be individually driven to zero by means of set zero signals (X set zero, on Figure 8-48) supplied by the discrete output logic of the computer. The set zero signal is supplied to the set zero control circuit which gates the 50 pps output from the fixed oscillator into the pulse counter, provided the display device counter is not already at zero. The pulses from the fixed oscillator then drive the motor in the normal manner until the counter is zeroed. The pulses are applied in such a manner that the count always decreases, regardless of the initial value.

Interfaces

The IVI interfaces, which are made with the computer and the IGS Power Supply, are described under the Interfaces heading in the DIGITAL COMPUTER SYSTEM OPERATION part of this section.

HORIZON SENSOR SYSTEM

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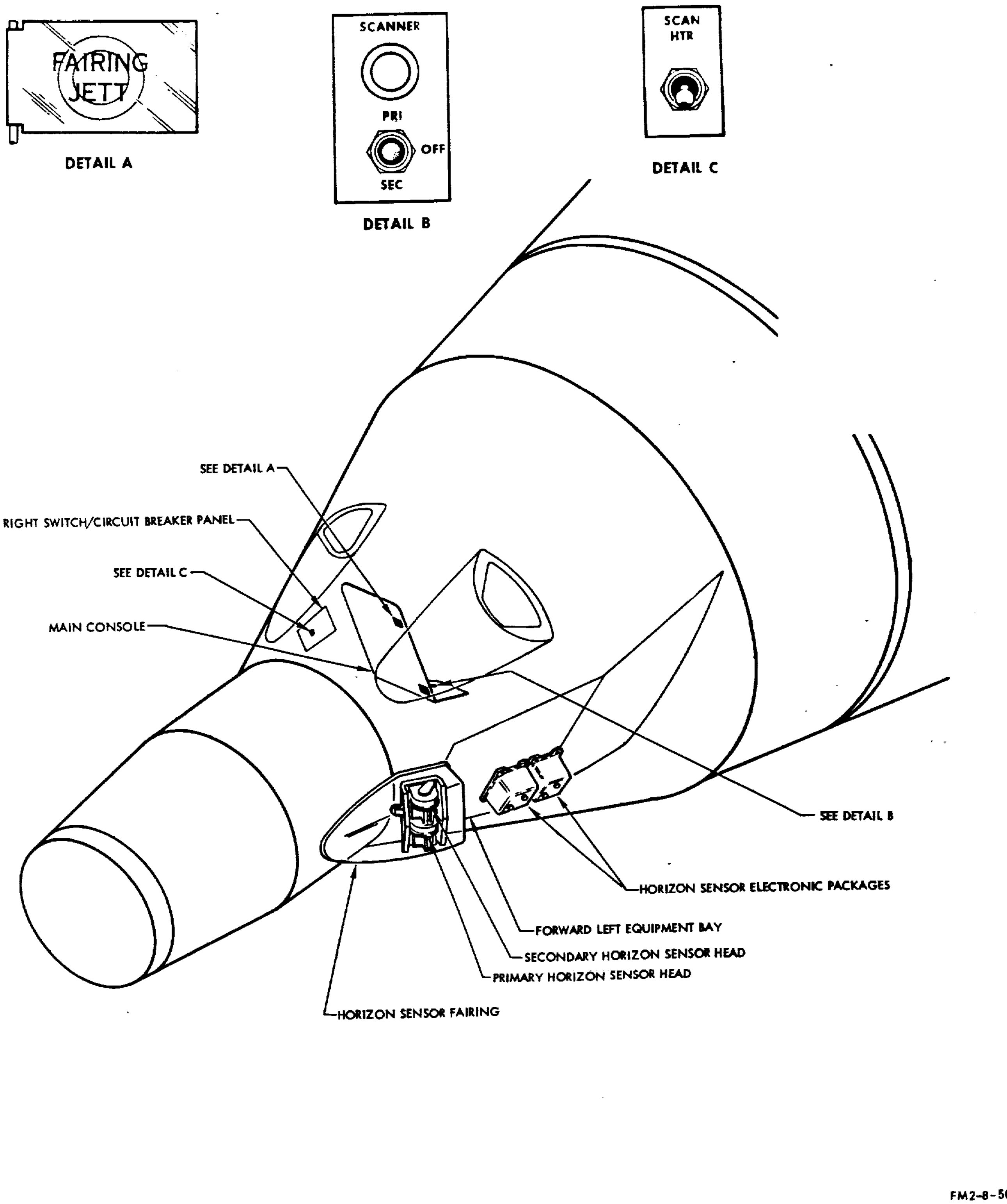


Figure 8-49 Horizon Sensor System

HORIZON SENSOR SYSTEM

SYSTEM DESCRIPTION

The Horizon Sensor System (Figure 8-49) consists of a sensor head, an electronics package and their associated controls and indicators. The system is used to establish a spacecraft attitude reference to earth local vertical and generates error signals proportional to the difference between spacecraft attitude and a horizontal attitude. Attitude error signals can be used to align either the spacecraft or the inertial platform to earth local vertical. The system has a null accuracy of 0.1 degree and is capable of operating at altitudes of 50 to 900 nautical miles. When the system is operating in the 50 to 550 nautical mile altitude range, measurable spacecraft attitude error is $\pm$ 14 degrees. When spacecraft attitude errors are between 14 and 20 degrees, the sensor output becomes non-linear but the direction of its slope always corresponds with the slope of the attitude error. When spacecraft attitude errors exceed 20 degrees, the system may lose track. Two complete systems are installed on the spacecraft. The second system is provided as a back-up in case of primary system malfunction.

SENSOR HEAD

The sensor head (Figure 8-50) contains equipment required to scan, detect and track the infrared gradient between earth and space, at the horizon. The sensor heads are mounted on the left side of the spacecraft and canted 14 degrees forward of the spacecraft pitch axis. Scanning is provided about the azimuth axis by a yoke assembly and about the elevation axis by a Positor (mirror positioning assembly). Infrared detection is provided by a bolometer and tracking by a servo loop which positions the Positor mirror.

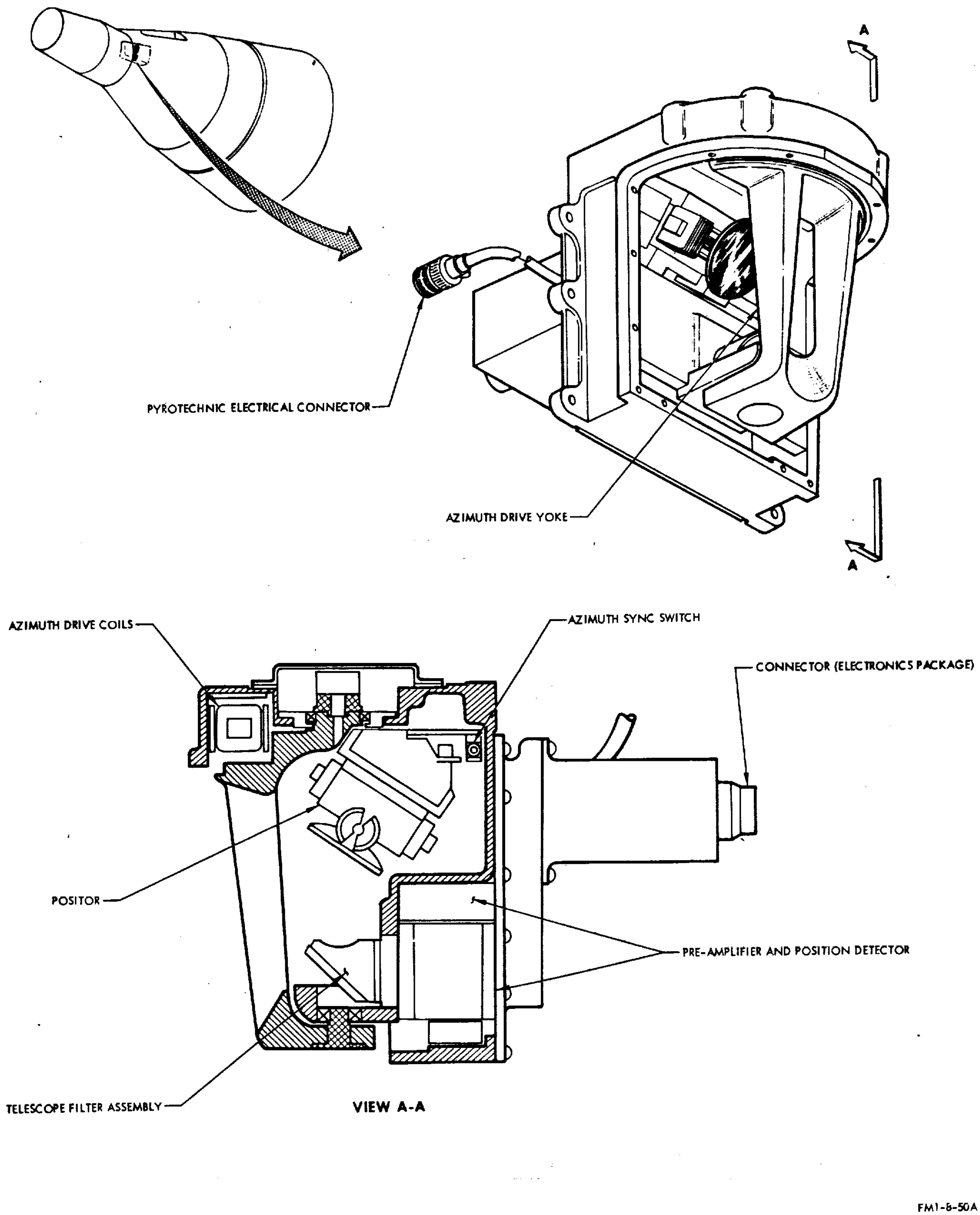


Figure 8-50 Horizon Sensor Scanner Head

ELECTRONICS PACKAGE

The electronics package (Figure 8-51) contains the circuitry required to provide azimuth and elevation drive signals to the sensor head and attitude error signals to spacecraft and platform control systems. Electrical signals from the sensor head, representing infrared radiation levels and optical direction, are used to generate elevation drive signals for the Positor. Signals are also generated to constantly drive the azimuth yoke from limit to limit. Attitude error signals are derived from the constantly changing Positor position signal when the system is tracking.

SYSTEM OPERATION

The primary Horizon Sensor System is energized during pre-launch by pilot initiation of the SCAN HTR and SCANNER PRI-OFF-SEC switches. Immediately after staging the pilot presses the JETT FAIR switch, exposing the sensor heads to infrared radiation. Initial acquisition time (the time required for the sensor to acquire and lock-on the horizon) is approximately 120 seconds; reacquisition time is approximately 10 seconds. The system can be used any time between staging and retro-section separation. At retro-section separation plus 80 milliseconds the sensor heads are automatically jettisoned, rendering the system inoperative.

Operation of the Horizon Sensor System depends on receiving, detecting and tracking the infrared radiation gradient between earth and space, at the horizon. To accomplish the above, the system employs infrared optics, infrared detection and three closely related servo loops. A functional block diagram of the Horizon Sensor System is provided in Figure 8-52.

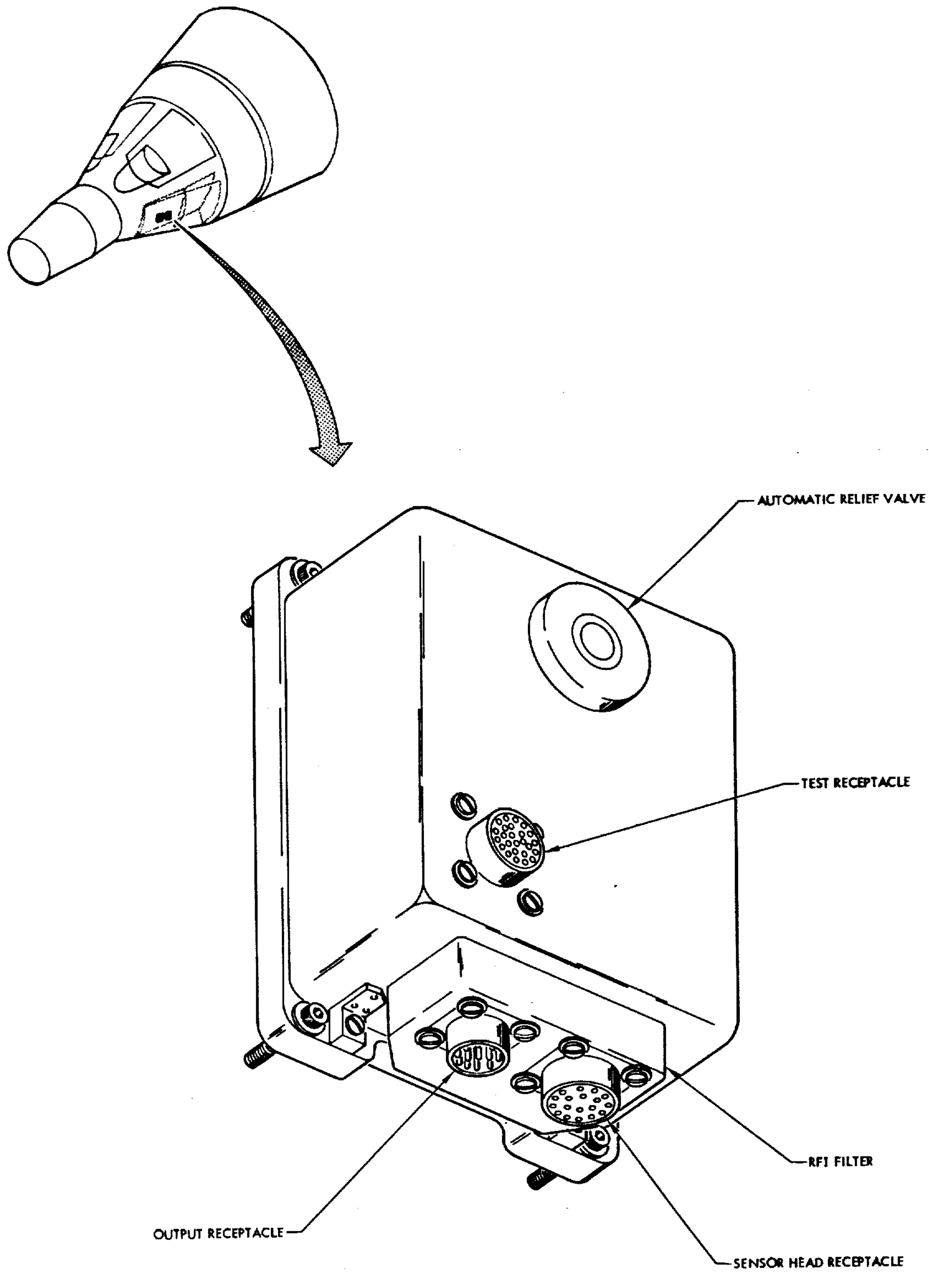


Figure 8-51 Horizon Sensor Electronic Package

FMG-64A

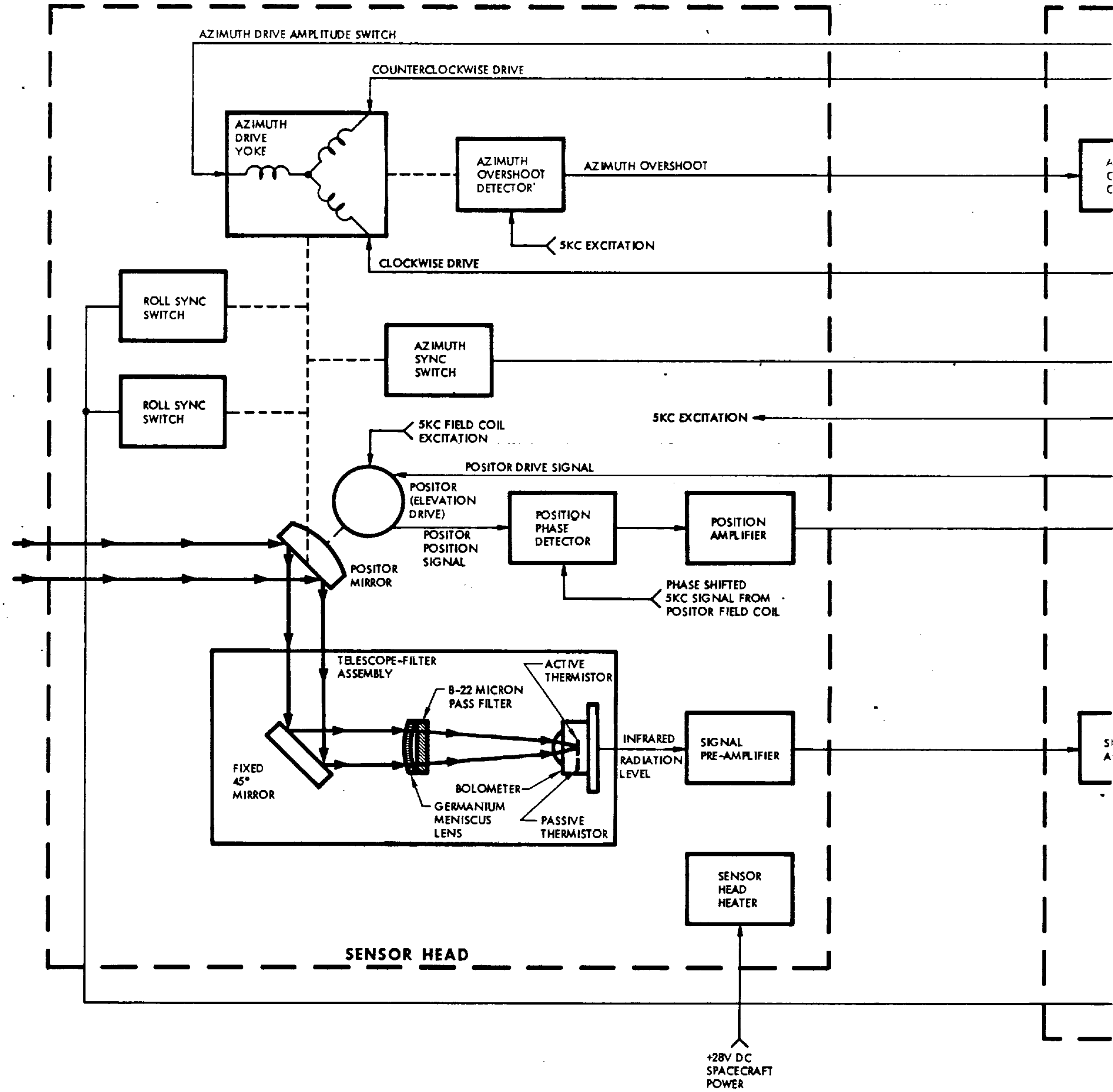
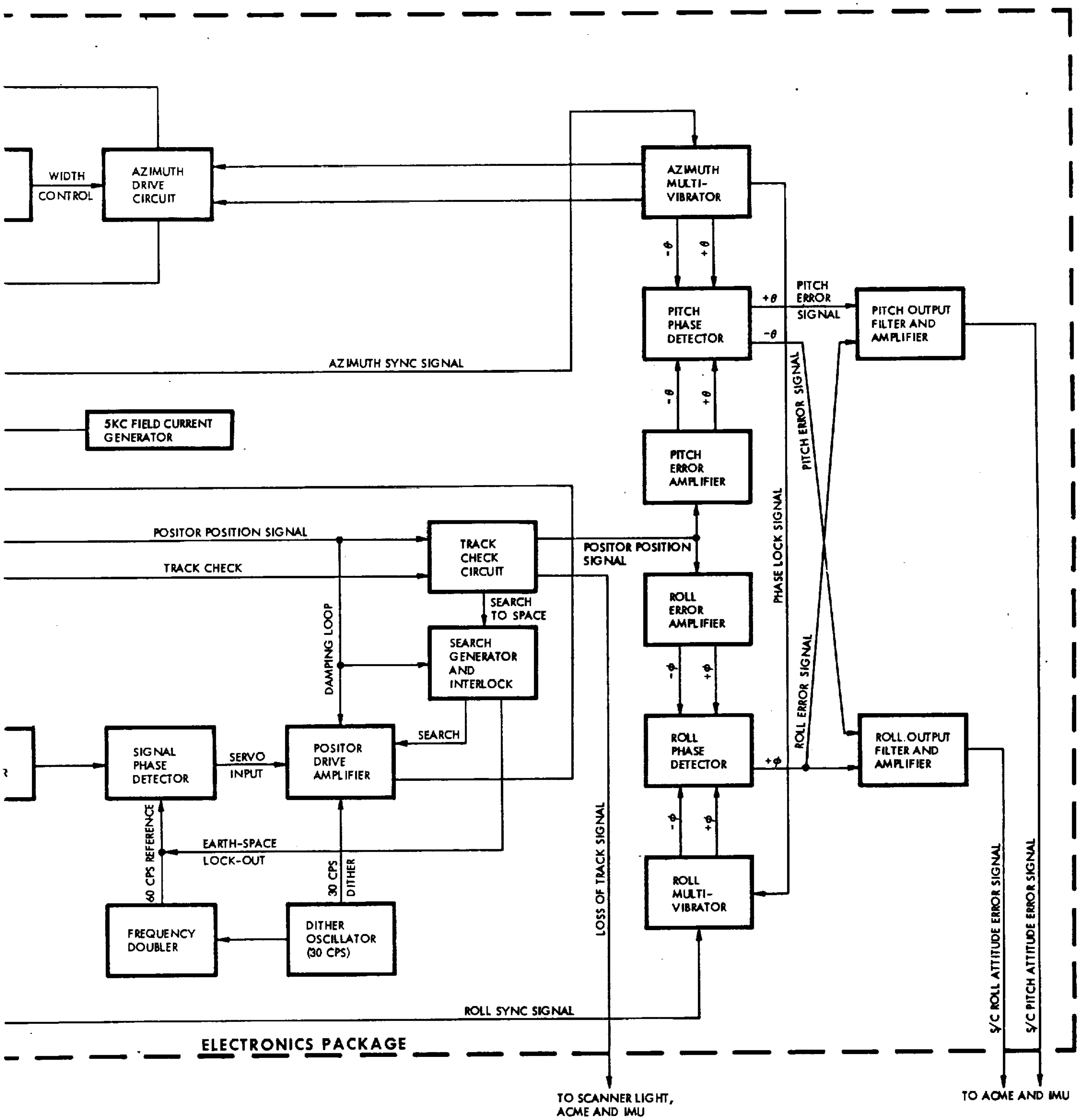


Figure 8-52 Horizon Sensor System Functional Block Diagram

INFRARED OPTICS

Infrared optics (Figure 8-53) consists of a Positor, a telescope-filter assembly and an azimuth drive yoke. All of these components are located in the sensor head. The Positor has a movable mirror which is used to position the system field of view about the horizon. Radiation is reflected by the Positor mirror into the telescope-filter assembly. A fixed mirror, in the telescope-filter assembly, directs infrared radiation into the telescope. The telescope-filter assembly contains a germanium meniscus objective lens, an infrared filter and a germanium-immersed thermistor bolometer. The objective lens is used to direct all the infrared radiation, reflected by the mirrors, on the germanium immersion lens of the bolometer. The infrared filter is used to eliminate radiation of undesired frequencies. The filter has a band pass of 8 to 22 microns (80,000 to 220,000 angstroms). The germanium immersion lens focuses the infrared radiation on an immersed thermistor.

The Horizon Sensor field of view is deflected through 160 degrees (± 80) in azimuth and 70 degrees (12 up and 58 down) in elevation by rotating the Positor mirror. The Positor is rotated in azimuth by a drive yoke. Rotation is about an axis which runs through the center of the infrared ray bundle on the surface of the Positor mirror. The yoke is driven at a one cycle per second rate by circuitry in the electronics package. The center of the azimuth scan is 14 degrees forward of the spacecraft pitch axis. This is due to the mounting of the scanner heads. Elevation deflection is provided by the Positor which tilts the Positor mirror as required to search for or track the horizon. The


WIDTH
CONTROL
AZIMUTH DRIVE CIRCUIT
AZIMUTH MULTI-VIBRATOR
PITCH PHASE DETECTOR
PITCH ERROR SIGNAL
PITCH OUTPUT FILTER AND AMPLIFIER
AZIMUTH SYNC SIGNAL
PITCH ERROR SIGNAL
5KC FIELD CURRENT GENERATOR
PITCH ERROR AMPLIFIER
PHASE LOCK SIGNAL
POSITOR POSITION SIGNAL
TRACK CHECK
TRACK CHECK CIRCUIT
POSITOR POSITION SIGNAL
ROLL ERROR AMPLIFIER
SEARCH TO SPACE
SEARCH GENERATOR AND INTERLOCK
DAMPING LOOP
ROLL ERROR SIGNAL
SEARCH
SIGNAL PHASE DETECTOR
SERVO INPUT
POSITOR DRIVE AMPLIFIER
ROLL PHASE DETECTOR
ROLL OUTPUT FILTER AND AMPLIFIER
60 CPS REFERENCE
EARTH-SPACE LOCK-OUT
30 CPS
DITHER
LOSS OF TRACK SIGNAL
ROLL MULTI-VIBRATOR
S/C ROLL ATTITUDE ERROR SIGNAL
S/C PITCH ATTITUDE ERROR SIGNAL
FREQUENCY DOUBLER
DITHER OSCILLATOR (30 CPS)
ROLL SYNC SIGNAL
ELECTRONICS PACKAGE
TO SCANNER LIGHT, ACME AND IMU
TO ACME AND IMU

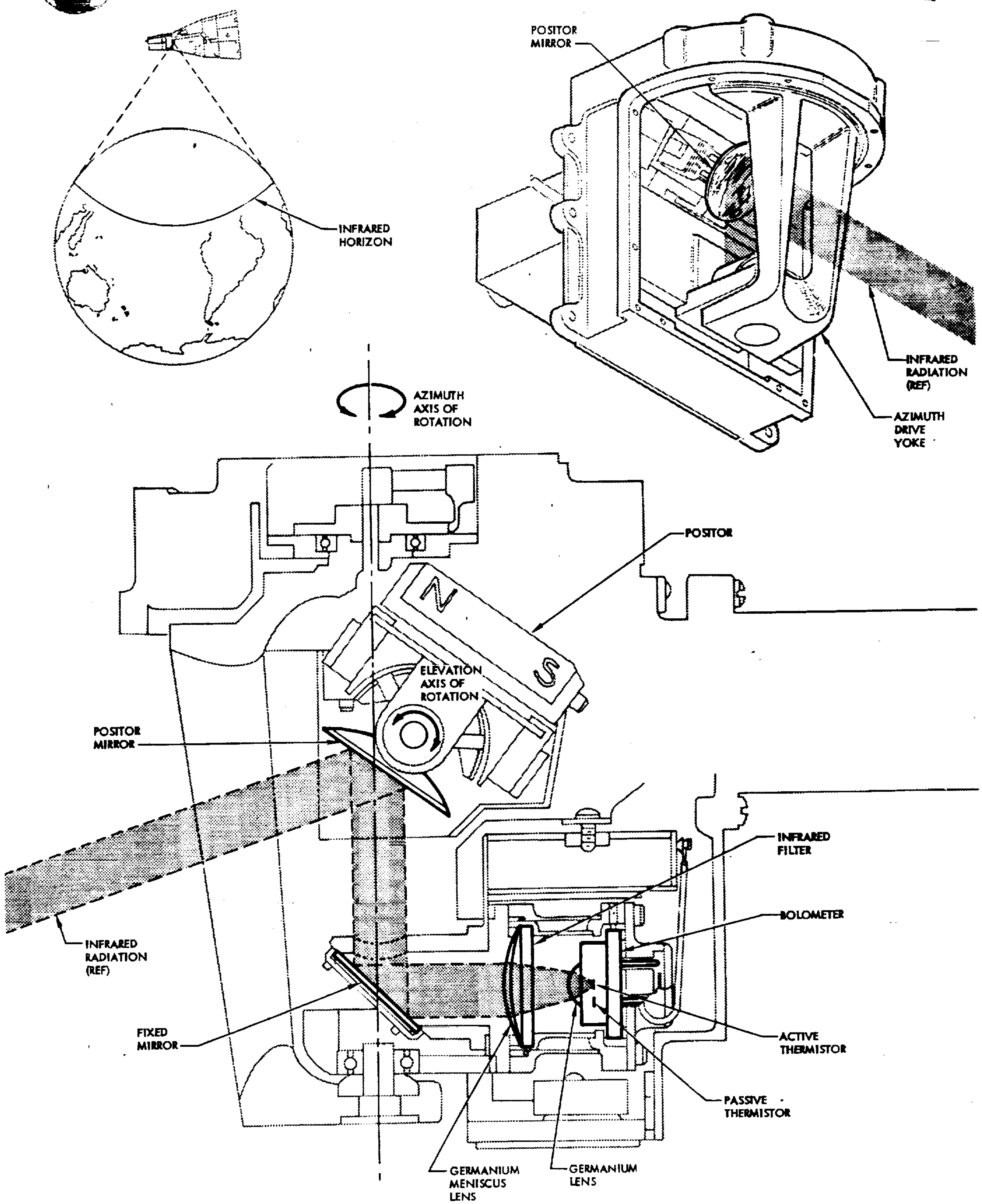


Figure 8-53 Infrared Optics

rate at which the Positor tilts the mirror is a function of the mode of operation (track or search). In search mode,the Positor mirror moves at a two cps search rate plus a 30 cps dither rate. In track mode,the Positor mirror moves at a 30 cps dither rate, plus, if there is any attitude error, a one or two cps track rate. The one or two cps track rate depends on the direction of spacecraft attitude error.

INFRARED DETECTION

Infrared radiation is detected by the germanium-immersed thermistor bolometer. The bolometer contains two thermistors (temperature sensitive resistors) which are part of a bridge circuit. One of the thermistors (active) is exposed to infrared radiation from the horizon. The second thermistor (passive) is located very near the first thermistor but it is separated from infrared radiation. Radiation from the horizon is sensed by the active thermistor which changes resistance and unbalances the bridge circuit. The unbalanced bridge produces an output voltage which is proportional to the intensity of the infrared radiation. If only one thermistor were used, the bridge would also sense temperature changes caused by conduction or convection; to prevent this, a passive (temperature reference) thermistor is used.

The passive thermistor changes resistance the same amount as the active thermistor, for a given ambient temperature change, keeping the bridge balanced. The passive thermistor is not exposed to infrared radiation and allows the bridge to become unbalanced when the active thermistor is struck by radiation from the horizon.

SERVO LOOPS

The three servo loops used by the Horizon Sensor System are: the track loop, the azimuth drive loop and the signal processing loop. Some of the circuitry is used by more than one servo loop and provides synchronization.

Track Loop

The track loop (Figure 8-54) is used to locate and track the earth horizon with respect to the elevation axis. Two modes of operation (search and track) are used in the track loop. The search mode is selected automatically when the system is first energized and used until the horizon is located. After the horizon is located and the signal built up to the required level, the track mode is automatically selected.

Search Mode

The search mode is automatically selected by the system any time the horizon is not in the field of view. The purpose of the search mode is to move the system line of sight through its elevation scan range until the horizon is located. (The system line of sight is moved by changing the angle of the positor mirror.) When the system is initially energized, the Positor position signal is used to turn on a search generator. The generator produces a two cps AC search voltage which is applied to a summing junction in the Positor drive amplifier. A second signal (30 cps dither) is also applied to the summing junction. (The dither signal is present any time the system is energized.) The search and dither voltages are summed and amplified to create a Positor drive signal. This drive signal is applied to the drive coils of the Positor causing it to tilt the Positor mirror. The dither portion of the signal causes the mirror to oscillate about its elevation axis at a 30 cps rate and

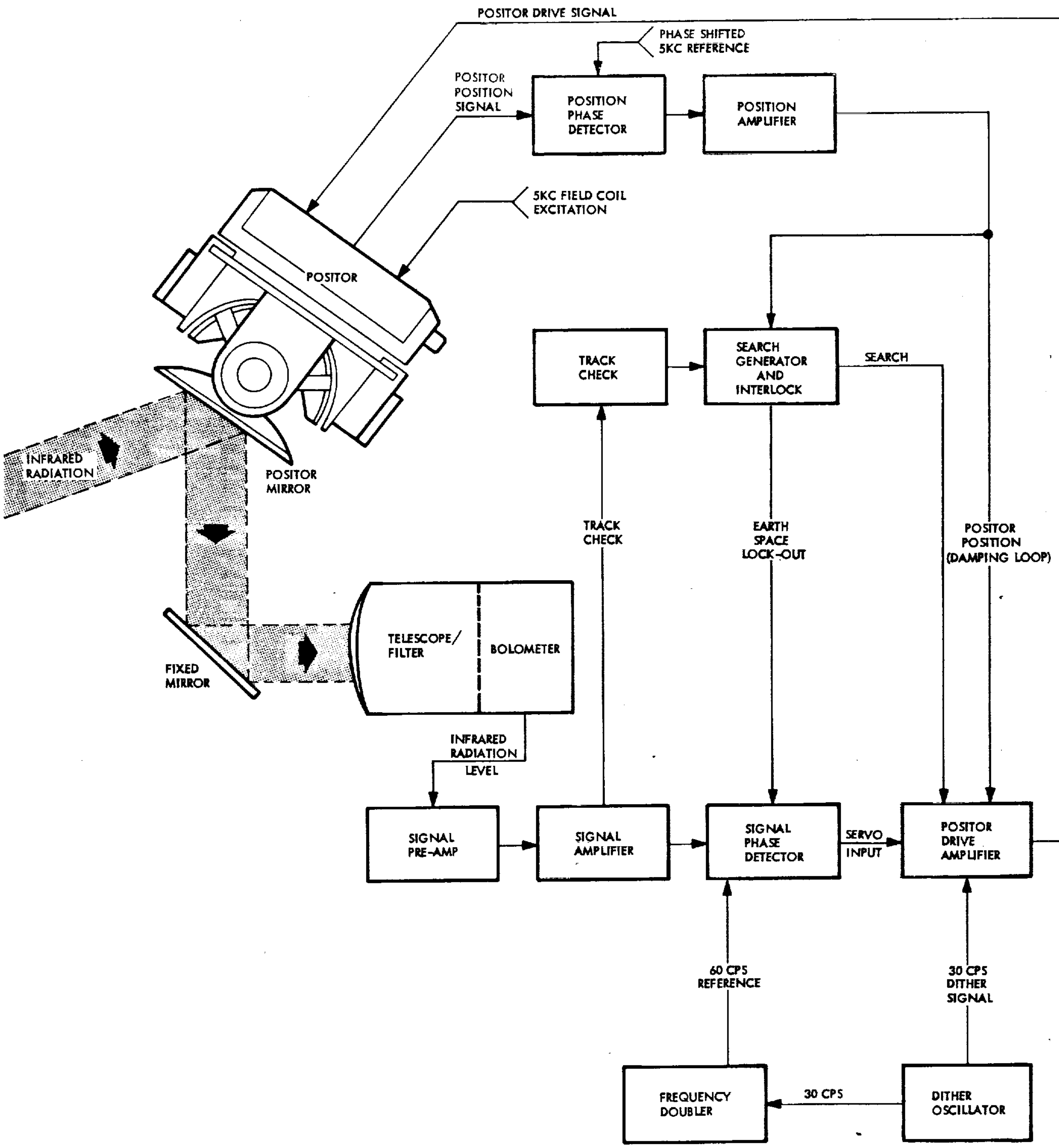


Figure 8-54 Track Loop Block Diagram

through an angle which represents approximately four degrees change in the line of sight. The search portion of the signal will drive the Positor mirror up to an angle which represents a line of sight 12 degrees above the spacecraft azimuth plane. During the up scan (earth to space) a lock-out signal is applied to the servo loop to prevent the system from locking on to false horizon indications. When the positive limit of the search voltage (12 degrees up) is reached, the voltage changes phase and the system begins to scan from 12 degrees up to 58 degrees down. During the down scan (space to earth), the lock-out signal is not used and the system is free to select track mode if the horizon comes within the field of view.

The bolometer output (indication of infrared radiation) is used to determine when the horizon comes within view and to initiate the track mode of operation. As the system line of sight crosses the horizon (from space to earth), a sharp increase in infrared radiation is detected by the bolometer. The bolometer bridge output now produces a 30 cps AC signal. (The 30 cps is caused by the dither signal driving the line of sight back and forth across the horizon.) The bolometer bridge output is amplified and applied to the track check circuit. When the 30 cps signal reaches the track check circuit, it causes a tracking relay to be energized indicating that the horizon is in the field of view. Contacts of the relay apply a bias to the search generator, turning it off and removing the search voltage from the Positor drive signal. This places the system in the track mode of operation.

Track Mode

The bolometer output signal is used to determine the direction of the horizon from the center of the system line of sight. A Positor drive voltage of the proper phase is then generated to move the system line of sight until the horizon is centered in the field of view. The bolometer output signal is phase detected with respect to a 60 cps reference signal. The 60 cps signal is obtained by doubling the frequency output of the dither oscillator. Since both signals (30 cps dither and 60 cps reference) come from the same source, the phase relationship should be a constant. However, when the horizon is not in the center of the field of view, the bolometer output is not symmetrical. The time required for one complete cycle is the same as for dither but the zero crossover is not equally spaced, in time, from the beginning and end of each cycle. The direction the zero crossover is shifted from center depends on whether the horizon is above or below the center of the field of view. The phase detector determines the direction of shift (if any) and produces DC pulses of the appropriate polarity. The output of the signal phase detector is applied to the Positor drive amplifier where it is summed with the dither signal. The composite signal is then amplified and used to drive the Positor mirror in the direction required to place the horizon in the center of the field of view.

A pickup coil, wound on the permanent magnet portion of the Positor drive mechanism, produces an output signal which is proportional (in phase and amplitude) to the position of the Positor mirror. This Positor position signal is phase detected to determine the actual position of the mirror. The detector output is then amplified and used for two purposes in the track loop: to activate the search generator when the tracking relay is de-energized and as a rate damping feedback to the Positor drive amplifier. (When the tracking relay is energized, it biases the search generator to cutoff.)

Azimuth Drive Loop

The azimuth drive loop (Figure 8-55) provides the drive voltage, overshoot control and synchronization required to move the system line of sight through a 160 degree scan angle at a one cps rate. The azimuth drive loop consists of an azimuth overshoot detector, azimuth control circuit, azimuth multivibrator, azimuth drive coils and an azimuth drive yoke.

Azimuth Overshoot Detector

The azimuth overshoot detector does not, as the name implies, detect the azimuth scan overshoot. It instead detects when the azimuth drive yoke reaches either end of its scan limit. The detector is a magnetic pickup, located near the azimuth drive yoke and excited by a 5 KC signal from the field current generator. Two iron slugs, mounted on the azimuth drive yoke, pass very near the magnetic pickup when the yoke reaches the scan limit. The slugs are positioned 160 degrees apart on the yoke to represent each end of the scan. When one of the iron slugs passes near the magnetic pickup, it changes the inductance and causes the 5 KC excitation signal to be modulated with a pulse. Since the azimuth scan rate is one cps and the modulation occurs at each end of the scan, the overshoot pulse occurs at a two pps rate. Output of the overshoot detector is applied to the azimuth control circuit.

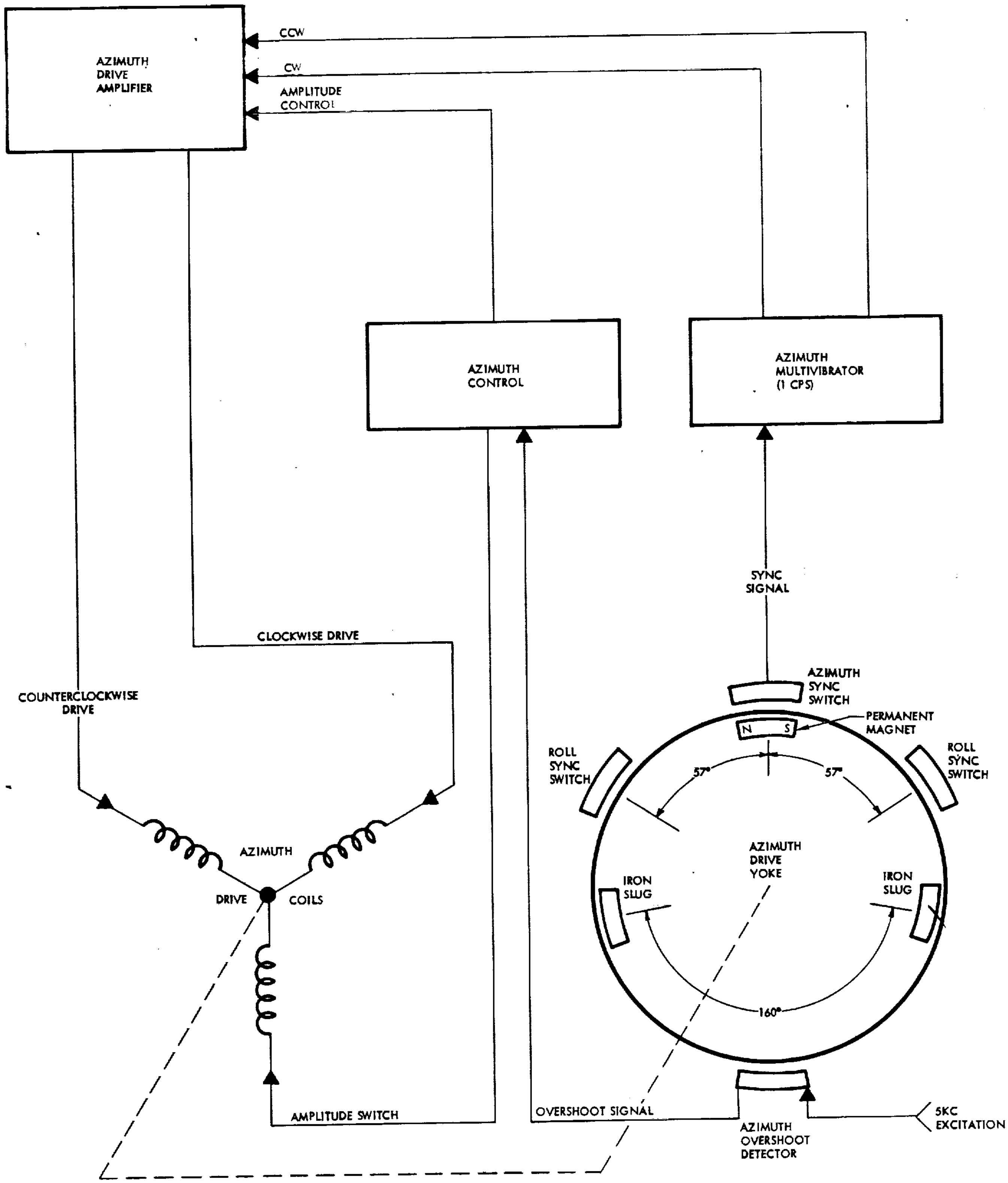


Figure 8-55 Azimuth Drive Loop Block Diagram

Azimuth Control Circuit

The azimuth control circuit generates two types of azimuth control voltages (coarse and fine) based on the azimuth overshoot signal. The azimuth overshoot detector output is rectified, filtered, peak detected and integrated to develop a DC control voltage proportional to the amplitude and width of the overshoot pulse. This control voltage serves two purposes: to provide continuous, fine control of the azimuth drive pulse and, when the control voltage reaches a high enough level (indicating a large overshoot), provide a coarse (step) control of the reference voltage on the azimuth drive coils. The fine control is obtained by applying the control voltage, as a bias, to the azimuth drive amplifier. The coarse control is obtained by energizing a relay, which switches the reference voltage on the azimuth drive coils when the control voltage reaches a high enough level. The level at which the relay energizes is determined by a zener diode which breaks down and biases a relay driver into conduction. The relay driver then energizes a relay which switches the DC voltage on the reference winding of the azimuth drive coils.

Azimuth Multivibrator

The azimuth multivibrator provides the direction control signal for the azimuth drive. The multivibrator is synchronized by pulses from the azimuth sync switch. The sync switch is located next to the azimuth drive yoke and is closed each time the yoke passes through the center of its 160 degree scan. The switch produces a two pps output which is used to switch the state of the multivibrator. The multivibrator then produces a one cps square wave signal which is synchronized with the motion of the azimuth drive yoke. The positive half of the square wave controls the azimuth drive in one direction and the negative half controls the drive in the other direction. Output of the multivibrator is applied to the azimuth drive amplifier.

Azimuth Drive Amplifier

The azimuth drive amplifier adjusts the width of multivibrator output pulses to control the azimuth drive yoke. The output pulse width, from the drive amplifier, depends on the amount of control voltage (bias) provided by the azimuth control circuit. When the amount of azimuth yoke overshoot is large, the control voltage is high and the output pulse width is narrow. As the amount of overshoot decreases, the control bias decreases and the output pulse width increases. This provides a continuous, fine control over the drive pulse and consequently the amount of azimuth drive yoke travel.

Azimuth Drive Coils

The azimuth drive coils convert drive signals into a magnetic force. The coils are mounted next to, and their magnetic force exerted on, the azimuth drive yoke. The direction of magnetic force is determined by which drive coil is energized.

Azimuth Drive Yoke

The azimuth drive yoke is a means of mechanically moving the system line of sight through a scan angle. (The Positor is mounted inside the azimuth drive yoke and the rotation is around the center line of the infrared ray bundle on the Positor mirror.) The azimuth drive yoke is spring loaded to its center position and the mass adjusted to give it a natural frequency of one cps. Mounted on the yoke are two iron slugs and a permanent magnet. The iron slugs are used in conjunction with the azimuth overshoot detector mentioned previously. The magnet is used to activate sync switches located next to the drive yoke. The switches synchronize mechanical motion of the yoke with electrical signals. The function of the azimuth sync switch was described in the azimuth multivibrator paragraph. The function of the two roll sync switches will be described in the phase detectors paragraph of the signal processing loop.

Signal Processing Loop

The signal processing loop (Figure 8-56) converts tracking and azimuth scan information into attitude error signals. (The error signals can be used to align the spacecraft and/or the Inertial Guidance System to the earth local vertical.) A complete servo loop is obtained by utilizing two other spacecraft systems (Attitude Control and Maneuver Electronics and the Propulsion System). Attitude error signals, generated by the Horizon Sensor System are used by the Attitude Control and Maneuver Electronics (ACME) (in the horizon scan mode) to select the appropriate thruster (or thrusters) and generate a fire command. The fire command causes the Propulsion System to produce thrust in the desired direction. As the thrust changes spacecraft attitude, in the appropriate direction, the attitude error signals decrease in amplitude. When the spacecraft attitude comes within preselected limits (0 to -10 degrees in pitch and $\pm$ 5 degrees in roll), as indicated by error signal amplitude, the ACME stops generating fire commands. As long as the spacecraft attitude remains within the preselected limits, it is allowed to drift freely. If the attitude exceeds the limits, thrust is automatically applied to correct the error.

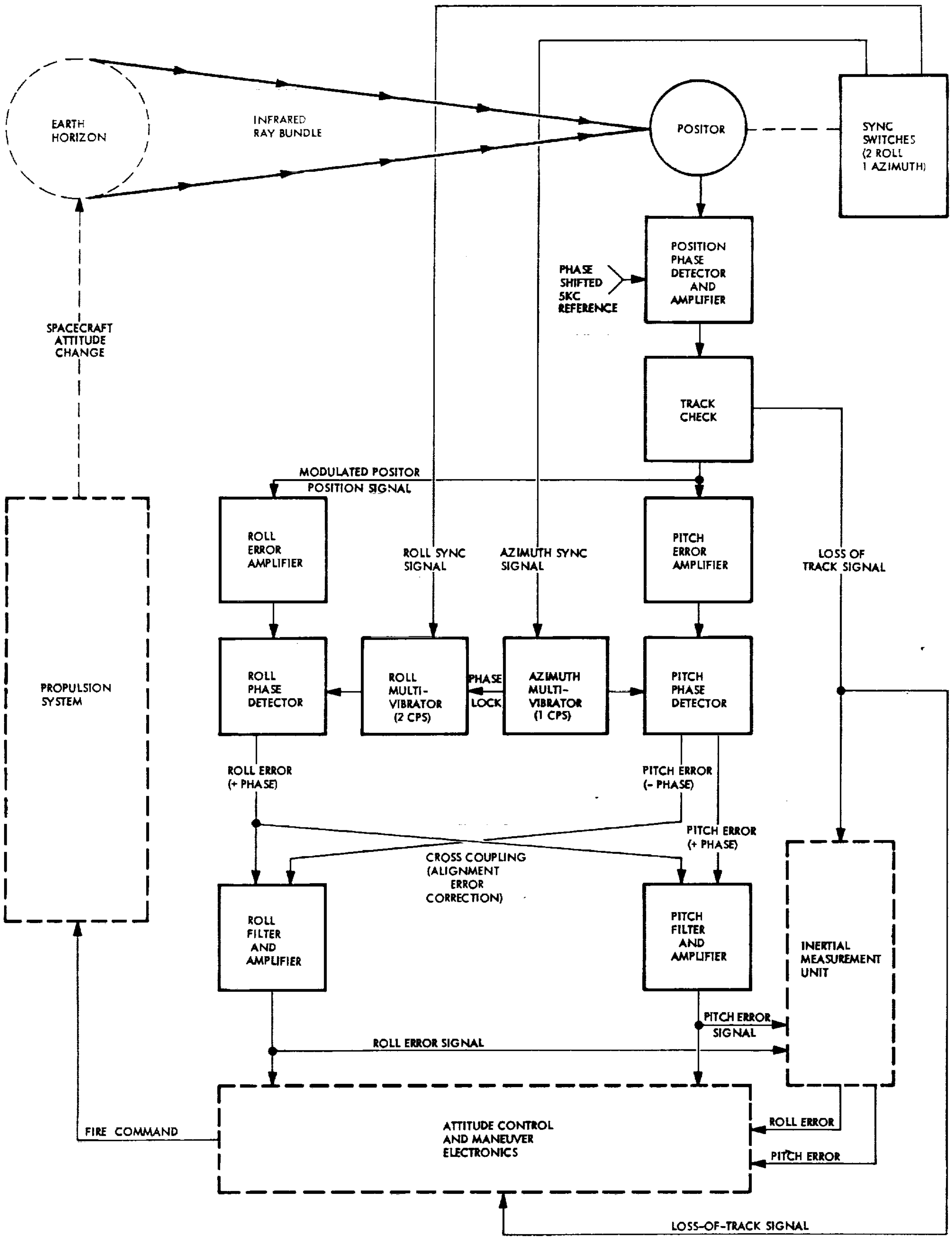


Figure 8-56 Signal Processing Loop Block Diagram

An indirect method of controlling spacecraft attitude (on spacecraft 7) with the Horizon Sensor involves a third spacecraft system (Inertial Guidance). This method can be used when it is desired to fine align the inertial measurement unit. Horizon Sensor attitude error signals are now used to continuously torque gyros in the inertial platform, aligning them to the local vertical. The platform attitude error signals are then used by the ACME (in the platform mode) to generate fire commands for the Propulsion System. Using this method of attitude control, the spacecraft is held to within ±1.1 degrees of the platform attitude in all three axes.

The inertial platform can also be aligned by the Horizon Sensor without using a servo loop. To align the platform without a closed servo loop, the pilot must manually maintain spacecraft attitude as near null as possible. (The Horizon Sensor attitude error signals are most accurate when the spacecraft is in a horizontal attitude with respect to the earth surface.) Attitude error signals are then used to torque platform gyros and have no direct effect on spacecraft attitude.

The Horizon Sensor System also provides a loss of track indication to both the ACME and Inertial Guidance System. The signal is used to prevent the ACME or platform from aligning to a false horizon. The loss of track signal is also used to illuminate the SCANNER light on the pedestal, informing the pilot that the system is not tracking. (Spacecraft attitude must be held within ±20 degrees of the horizontal for the system to track.)

Tracking Geometry

Horizon Sensor tracking geometry (Figure 8-57) is composed of the elevation angles (θ) generated by the track loop and the azimuth angles (ϕ) generated by the azimuth drive loop. Angles are compared in time and phase to generate an error signal proportional to the elevation angle change with respect to the azimuth scan angle.

As explained in the track loop paragraph, the system will lock on in elevation and track the earth horizon. A dither signal causes the Positor to move the system line of sight about the horizon at a 30 cps rate. The track loop will move the Positor mirror such that the horizon is always in the center of the dither pattern. It was also explained, in the azimuth drive loop paragraph, that the system line of sight is continuously moved through a 160 degree azimuth scan angle at a one cps rate.

When the spacecraft is in a horizontal attitude, the azimuth scan has no effect on the elevation angle of the Positor as it tracks the horizon. If the spacecraft is in a pitch up attitude, the elevation angle (θ) will decrease as the azimuth angle (ϕ) approaches 80 degrees forward and increase as angle ϕ approaches 80 degrees aft. If the spacecraft is in a pitch down attitude, the elevation angle will increase as the azimuth angle approaches 80 degrees forward and decrease as the azimuth angle approaches 80 degrees aft. This produces a one cps pitch error signal which is superimposed on the 30 cps Positor dither.

If the spacecraft has a roll right attitude, the elevation angle will increase as the azimuth angle approaches either limit and decrease as the azimuth angle approaches zero from either limit. If the spacecraft is in a roll left attitude

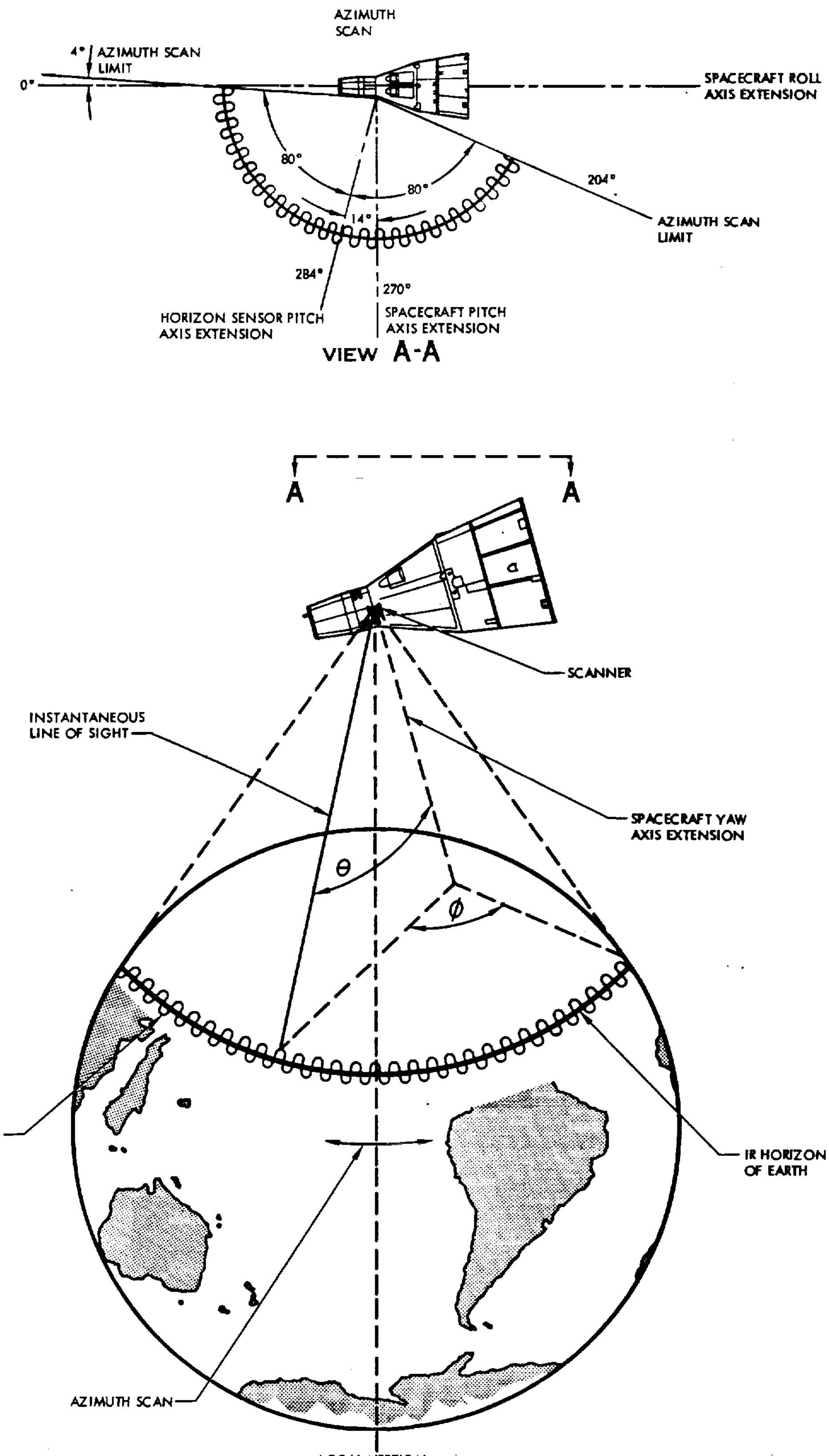


Figure 8-57 Horizon Sensor Tracking Geometry

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the elevation angle will decrease as the azimuth angle approaches either limit and increase as the azimuth angle approaches zero from either limit. This produces a two cps error signal which is superimposed on the 30 cps Positor dither.

Position Phase Detector

The Positor position phase detector compares the Positor pickoff signal with a 5 KC reference to determine the angle of the Positor mirror. (The mirror angle will be changing at a 30 cps dither rate, plus, if there is any spacecraft attitude error, a one and/or two cps error signal rate.) The phase detector output is then amplified and applied to the track check circuit.

Track Check

The track check circuit determines when the horizon is in the field of view. If the horizon is in the field of view,the track check circuit energizes a relay. Contacts of this relay connect the Positor position signal to the pitch and roll error amplifiers. A second relay in the track check circuit, energized when the system is not tracking, provides a loss of track indication to the inertial measurement unit and the ACME. The loss of track signal is 28 volts DC obtained through the ATT IND CNTL-LDG circuit breaker and switched by the track check circuit.

Error Amplifiers

In order to obtain individual pitch and roll attitude error outputs, error signal separation must be accomplished. This function is performed by two error amplifiers. The Positor position signal input to the error amplifiers is a composite 30 cps dither, one cps pitch error and two cps roll error signal. The pitch error amplifier is tuned to one cps and selects the pitch error signal only for amplification. The roll error amplifier is tuned to two cps and selects the roll error signal only for amplification. Each amplifier then amplifies and inverts their respective signals, producing two outputs each. The outputs are 180 degrees out of phase and of the same frequency as their input circuits were tuned. Output of each error amplifier is coupled to its respective phase detector.

Phase Detectors

Phase detectors compare the phase of pitch and roll error signals with one and two cps multivibrator reference signals to determine the direction of attitude error. The multivibrators are synchronized with motion of the azimuth drive yoke by three sync switches. Two sync switches, located at 57 degrees on either side of the center position of the yoke, synchronize the roll multivibrator with the motion of the yoke and set its frequency at two cps. The sync switches close each time the yoke passes, in either direction, producing four pulses for each cycle of the yoke. Each time a pulse is produced it changes the state of the multivibrator resulting in a two cps output. The azimuth multivibrator operates in the same manner except that it only has one sync switch, located at the center of the drive yoke scan, resulting in a one cps output frequency. The azimuth multivibrator also provides a phase lock signal to the roll multivibrator to assure not only frequency synchronization but correct phasing as well. The phase detectors themselves are actually reed relays, two for each detector, which are energized alternately by their respective multivibrator output signals. Contacts of these relays combine the two input signals in such a manner that two full wave rectified output signals are produced. The polarity of these pulsating DC outputs indicates the direction and the amplitude indicates the amount of attitude error about the Horizon Sensor pitch and roll axes. Since the sensor head was mounted at a 14 degree angle with respect to the spacecraft, the mounting error must be compensated for. Electrical rotation of the Horizon Sensor axes, to correspond with spacecraft axes, is accomplished by cross coupling a portion of the pitch and roll error signals.

Output Amplifier and Filter

The output amplifier-filter removes most of the two and four cps ripple from the rectified attitude error signals and amplifies the signals to the required level. The identical pitch and roll operational amplifiers, used as output stages for the Horizon Sensor System, are highly stable and have a low frequency response. The output signal amplitude is four tenths of a volt for each degree of spacecraft attitude error. The signals are supplied to the ACME for spacecraft alignment and to the inertial measurement unit for platform alignment.

HORIZON SENSOR POWER

Horizon Sensor power (Figure 8-58) is obtained from the 28 volt DC spacecraft bus and the 26 volt AC, 400 cps ACME power. The 28 volt DC power, supplied through the SCAN HTR switch, is used to maintain temperature in the sensor head and as power for the SCANNER lamp. Sensor head heaters are thermostatically controlled and operate any time the SCAN HTR switch is on. The 26 volt AC, 400 cps ACME power is provided by either the IGS or ACME inverter, depending on the position of the AC POWER selector. The 26 volt AC is used to produce seven different levels of DC voltage used in the Horizon Sensor. One of the voltages (31 volts DC) is obtained by rectifying and filtering the 26 volt AC input. The remaining six levels are obtained by transforming the 26 volts to the desired level, then rectifying, filtering and regulating it as required. The minus 27 volts DC output is used to excite one side of the bolometer bridge. The other side of the bridge is excited

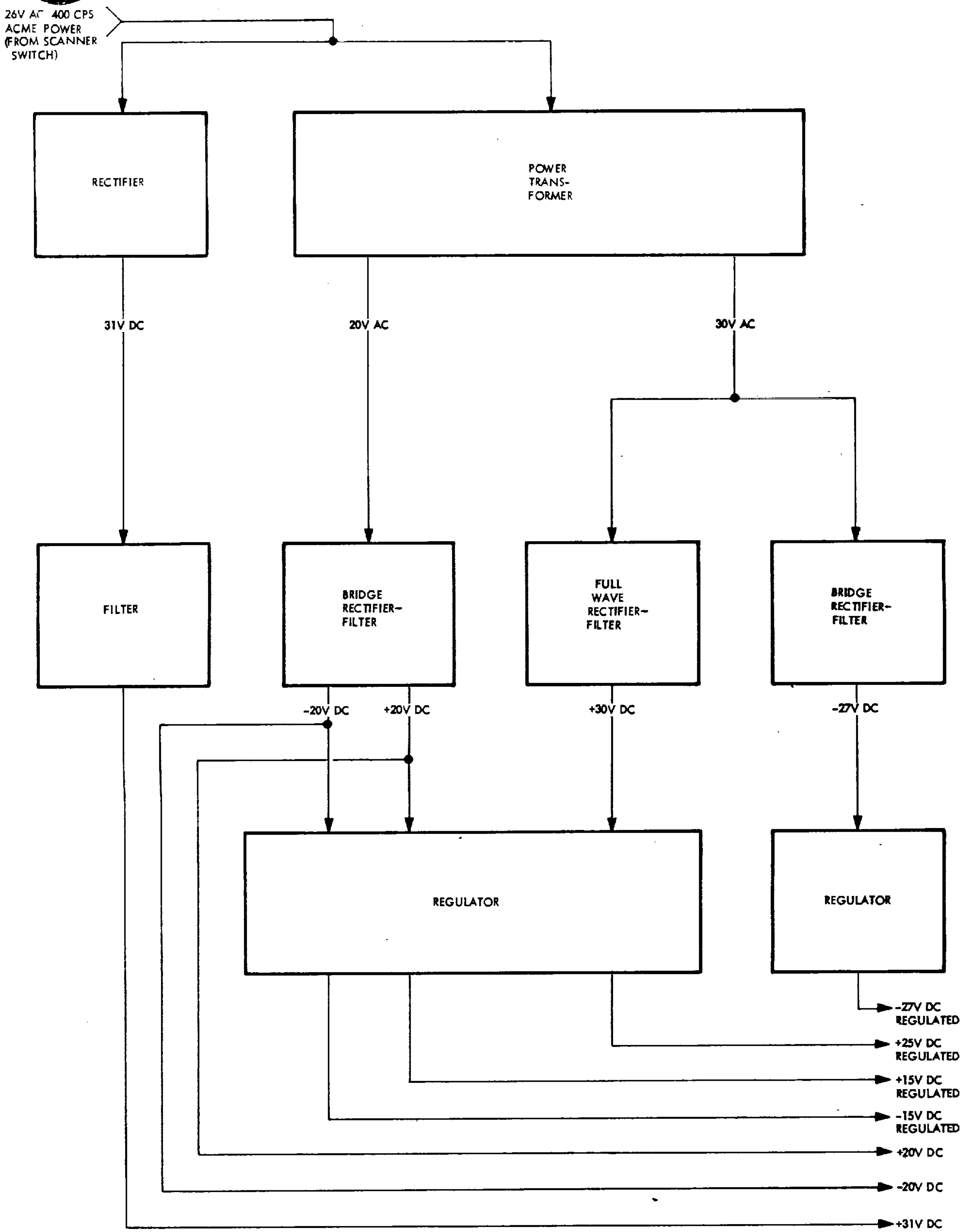

Figure 8-58 Horizon Sensor Power Supply Block Diagram

by plus 25 volts DC. Plus 25 volts DC is also used for transistor power in the error amplifiers. The 31 volt DC unregulated output is used as excitation for the azimuth drive yoke. The remaining four voltages (+15, -15, +20, -20) are all used for transistor power in the various electronic modules.

SYSTEM UNITS

The Horizon Sensor System (Figure 8-49) consists of two major units and five minor units. The minor units are: three switches, an indicator light and a fiberglass fairing. The three switches are mounted on the control panels for pilot actuation. The indicator light is mounted on the pedestal and, when illuminated, indicates a loss of track. The fiberglass fairing is dust proof and designed to protect the sensor heads, which it covers, from accidental ground damage or thermal damage during launch. The two major units are: the sensor head and the electronics package.

SENSOR HEAD

The sensor head (Figure 8-50) is constructed from a magnesium casting and contains a Positor, a telescope-filter assembly, a signal preamplifier, a position detector, an active filter and an azimuth drive yoke. The Positor (Figure 8-59) is a mirror positioning assembly designed to position a mirror about its elevation axis. The mirror is polished beryllium and is pivoted on ball bearings by a magnetic drive. The Positor also includes a position pickoff coil for determining the angle of the Positor mirror.

The telescope-filter assembly (see Figure 8-53) contains a fixed mirror, a germanium meniscus lens, an infrared filter and a germanium immersed thermistor bolometer. The fixed mirror is set at a 45 degree angle to reflect radiation from the

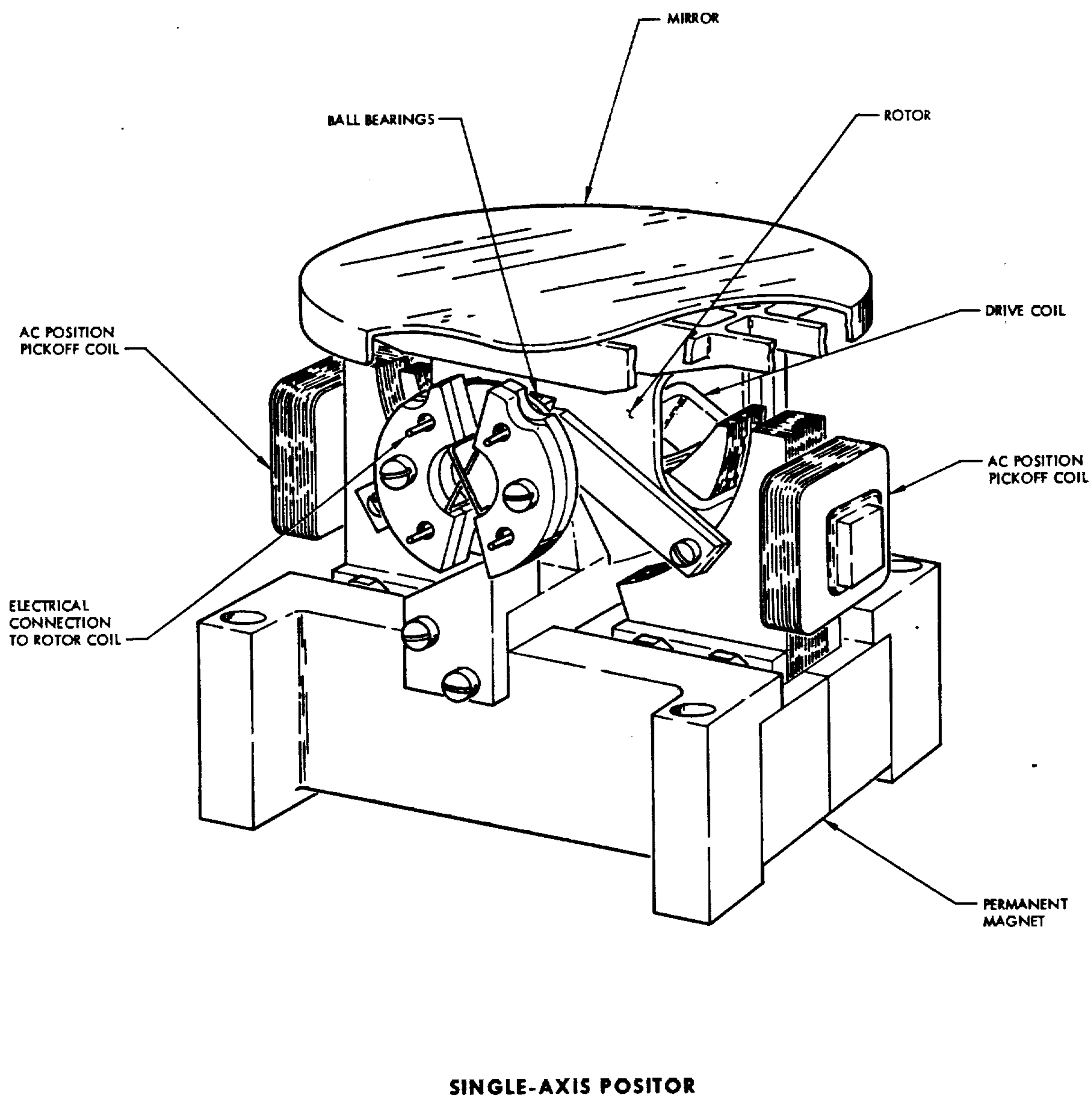

Figure 8-59 Horizon Sensor Single-Axis Positor

Positor mirror into the telescope. The germanium meniscus objective lens of the telescope is designed to focus incoming infrared radiation on the bolometer. The infrared filter, located immediately behind the objective lens, is designed to pass infrared radiation in the 8 to 22 micron range. The germanium immersed thermistor bolometer contains a culminating lens and two thermistors. The culminating lens directs all incoming radiation on one of the thermistors. The two thermistors are bonded to the rear of, and effectively immersed in, the culminating lens. Both thermistors are identical; however, one of the thermistors (active) is located at the focal point of the culminating lens. The other thermistor (passive) is located to one side of the focal point. The passive thermistor is used as an ambient temperature reference and does not react to direct infrared radiation.

Signal Preamplifier

The signal preamplifier is a low noise, high gain, four stage, direct coupled transistor amplifier. The preamplifier is made in modular form and potted in epoxy for thermal conductivity and protection from shock and vibration.

Position Detector

The position detector is a five KC phase detector designed to determine the position of the Positor mirror. The circuit produces a voltage which is proportional to the angle of the Positor mirror. Output of the detector is a DC voltage which varies at the same rate as the Positor mirror moves. The detector is made in modular form and potted in epoxy for thermal conductivity and protection from shock and vibration.

Azimuth Drive Yoke

The azimuth drive yoke provides a means of moving the Positor mirror about its azimuth axis. The yoke is magnetically driven and pivots on ball bearings. The Positor is mounted inside the azimuth drive yoke and is rotated through an azimuth scan angle of 160 (±80) degrees by the yoke. The azimuth axis of rotation is through the center line of the infrared ray bundle on the surface of the Positor mirror. Drive coils, located directly in front of the yoke, supply magnetic impulses to drive the yoke. Mounted on the edge of the yoke (see Figure 8-55) are two iron slugs and a permanent magnet. The iron slugs are used to induce an overshoot signal in the azimuth overshoot detector. The permanent magnet is used to synchronously close contacts on three sync switches, mounted around the periphery of the yoke.

ELECTRONICS PACKAGE

The electronics package (Figure 8-51) contains the circuitry necessary to control the azimuth yoke and Positor in the sensor head, as well as process attitude error signals. The package also contains a DC power supply and a five KC field current generator. The solid state circuitry is made in modular form and potted in epoxy for thermal conductivity and protection from shock and vibration.

TIME REFERENCE SYSTEM

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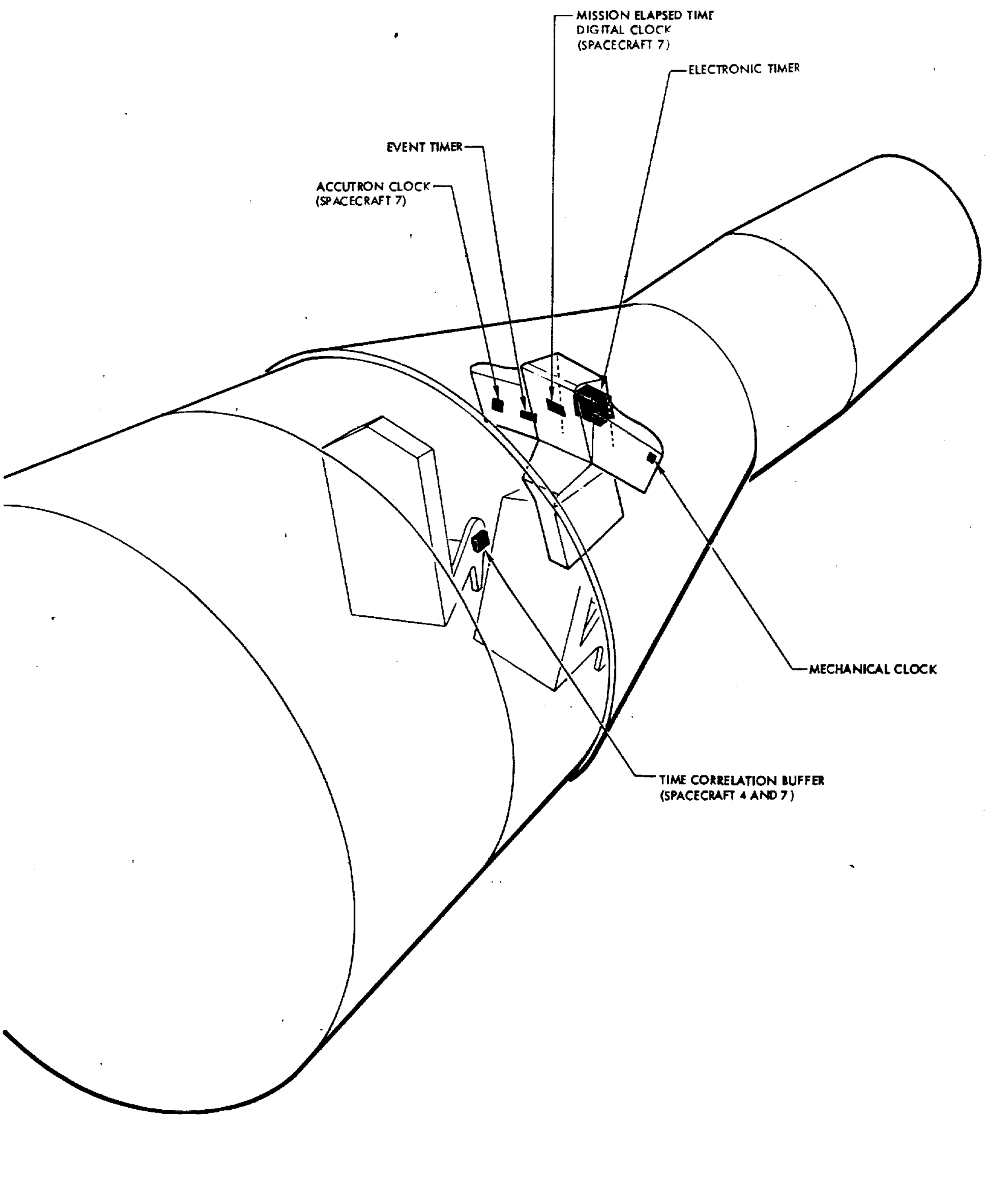


Figure 8-60 Time Reference System Equipment Locations

FM1-8-60B

TIME REFERENCE SYSTEM

SYSTEM DESCRIPTION

The Time Reference System (TRS) (Figure 8-60) provides the facilities for performing all timing functions aboard the spacecraft. The system is comprised of an electronic timer, a time correlation buffer, a mission elapsed time digital clock, an event timer, an Accutron clock and a mechanical clock. The event timer, mission elapsed time digital clock, Accutron clock and mechanical clock are all mounted on the spacecraft instrument panels. The electronic timer is located in the area behind the center instrument panel and the time correlation buffer is located in back of the pilot's seat.

The electronic timer provides (1) an accurate countdown of time-to-go to retrofire (TTG to T_R) and time-to-go to equipment reset (TTG to T_X), (2) time correlation for the PCM data system (Instrumentation) and the bio-med tape recorders, and (3) a record of elapsed time (ET) from lift-off.

The Time Correlation Buffer (TCB), used on spacecraft (S/C) 4 and 7, conditions certain output signals from the electronic timer, making them compatible with bio-med and voice tape recorders. Provision is included to supply buffered signals for Department of Defense (DOD) experiments if required.

The mission elapsed time digital clock (on S/C 7) provides a digital indication of elapsed time from lift-off. The digital clock counts pulses from the electronic timer and is therefore started and stopped by operation of the electronic timer.

The event timer provides the facilities for timing various short-term functions aboard the spacecraft. It is also started at lift-off to provide the pilots with a visual display of ET during the ascent phase of the mission. In case the electronic timer should fail, the event timer may serve as a back-up method of timing out T_R.

The Accutron clock (on S/C 4 and 7) provides an indication of Greenwich Mean Time (GMT) for the command pilot. The clock is powered by an internal battery and is independent of external power or signals.

The mechanical clock provides the pilot with an indication of GMT and the calendar date. In addition, it has a stopwatch capability. The stopwatch provides an emergency method of performing the functions of the event timer.

SYSTEM OPERATION

Four components of the Time Reference System (electronic timer, event timer, Accutron clock and mechanical clock) function independently of each other. The two remaining components (mission elapsed time digital clock and time correlation buffer) are dependent on output signals from the electronic timer. A functional diagram of the Time Reference System is provided in Figure 8-61.

The electronic timer, mission elapsed time digital clock, Accutron clock and the time-of-day portion of the mechanical clock operate continuously, during the spacecraft mission. The mechanical clock and Accutron clock are started during the pre-launch period. The electronic timer starts operating upon receipt of a remote start signal from the Sequential System at the time of lift-off.

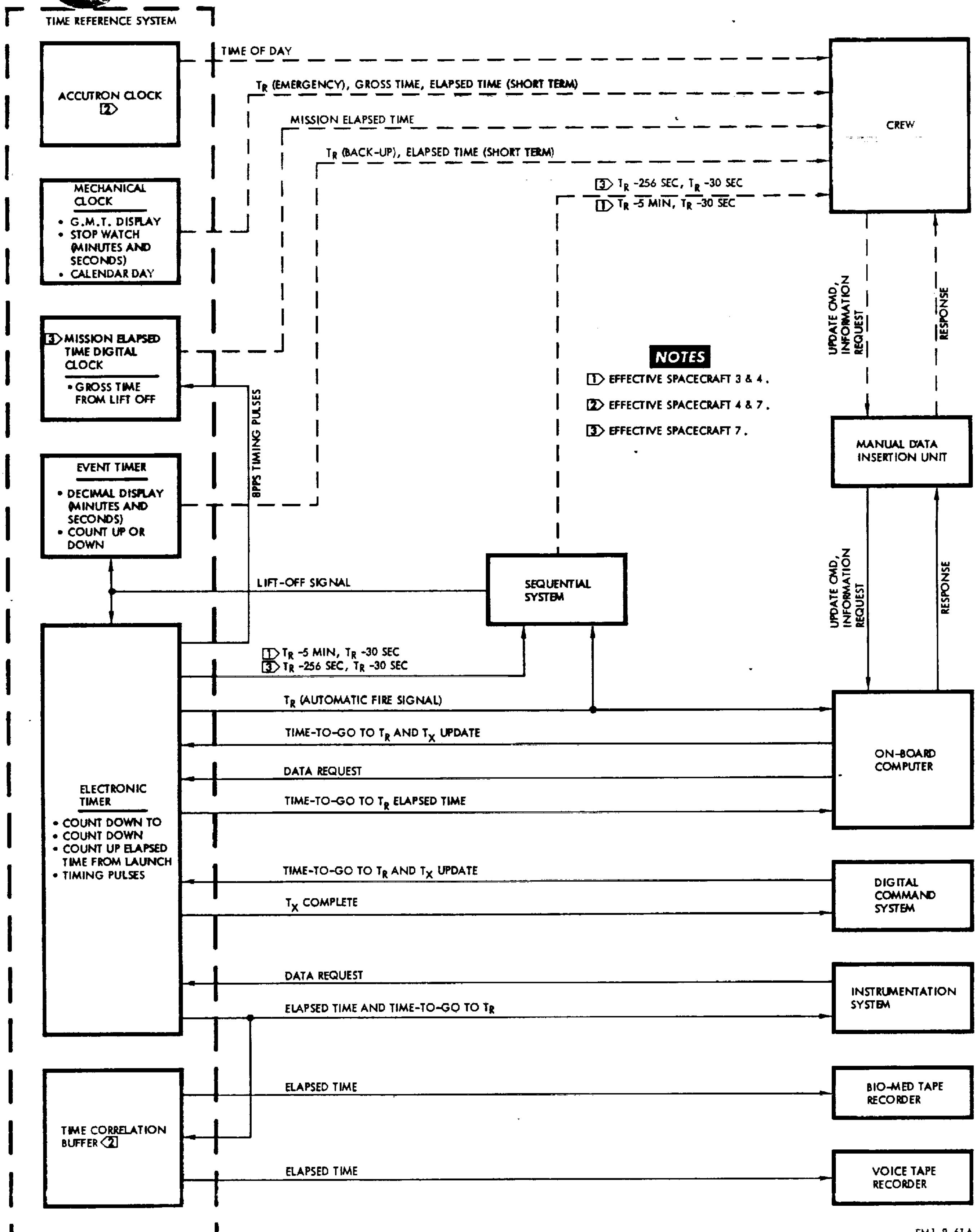


Figure 8-61 Time Reference System Functional Diagram

If the lift-off signal is not received from the Sequential System, the electronic timer can be started by actuation of the START-UP switch on the event timer. The mission elapsed time digital clock and time correlation buffer start operating upon receipt of output signals from the electronic timer.

During the mission, the event timer, Accutron clock and the stopwatch portion of the mechanical clock can be started and stopped, manually, at the descretion of the crew. At lift-off, however, the event timer is started by a remote signal from the Sequential System.

ELECTRONIC TIMER

General

At the time of lift-off, the electronic timer begins its processes of counting up elapsed time and counting down TTG to T_R and TTG to T_X. ET is counted up from zero to a maximum of approximately 24 days. The retrofire and equipment reset functions are counted down to zero from certain values of time which are written into the timer prior to lift-off. The timer is capable of counting TTG to T_R from a maximum of 24 days and to equipment reset from a maximum of two hours.

The TTG to T_R data contained by the timer may be updated at any time during the mission by insertion of new data. Updating may be accomplished either by a ground station, through the Digital Command System (DCS), or by the crew, via the Manual Data Insertion Unit (MDIU) and the digital computer. To prevent inadvertent, premature countdown of T_R as a result of equipment failure or personnel error during update, the timer will not accept any new time-to-go of less than 128 seconds duration on S/C 7 or 512 seconds on S/C 3 and 4. Upon receipt of new data of less than the inhibit time mertioned above, the timer will cause itself to be loaded with a time in excess of two weeks.

The TTG to T_X function of the timer serves to reset certain equipment which operates while the spacecraft is passing over a ground station equipped with telemetry. As the spacecraft comes within range, the ground station inserts, via the DCS, a TTG to T_X in the timer. Then, as the spacecraft moves out of the range of the ground station, the TTG to T_X reaches zero, and the equipment is automatically reset. If the ground station is unable to insert the time data, it may be done by the crew, using the MDIU and digital computer.

Information from the electronic timer is not continuously displayed; however, confirmation of satisfactory operation may be made by the readout of T_R data through use of the digital computer MDIU display readout capability.

NOTE

The mission elapsed time digital clock counts pulses from the electronic timer and, assuming no loss of pulses, will indicate the elapsed time recorded in the electronic timer. The digital clock does not, however, read out the elapsed time word from the electronic timer.

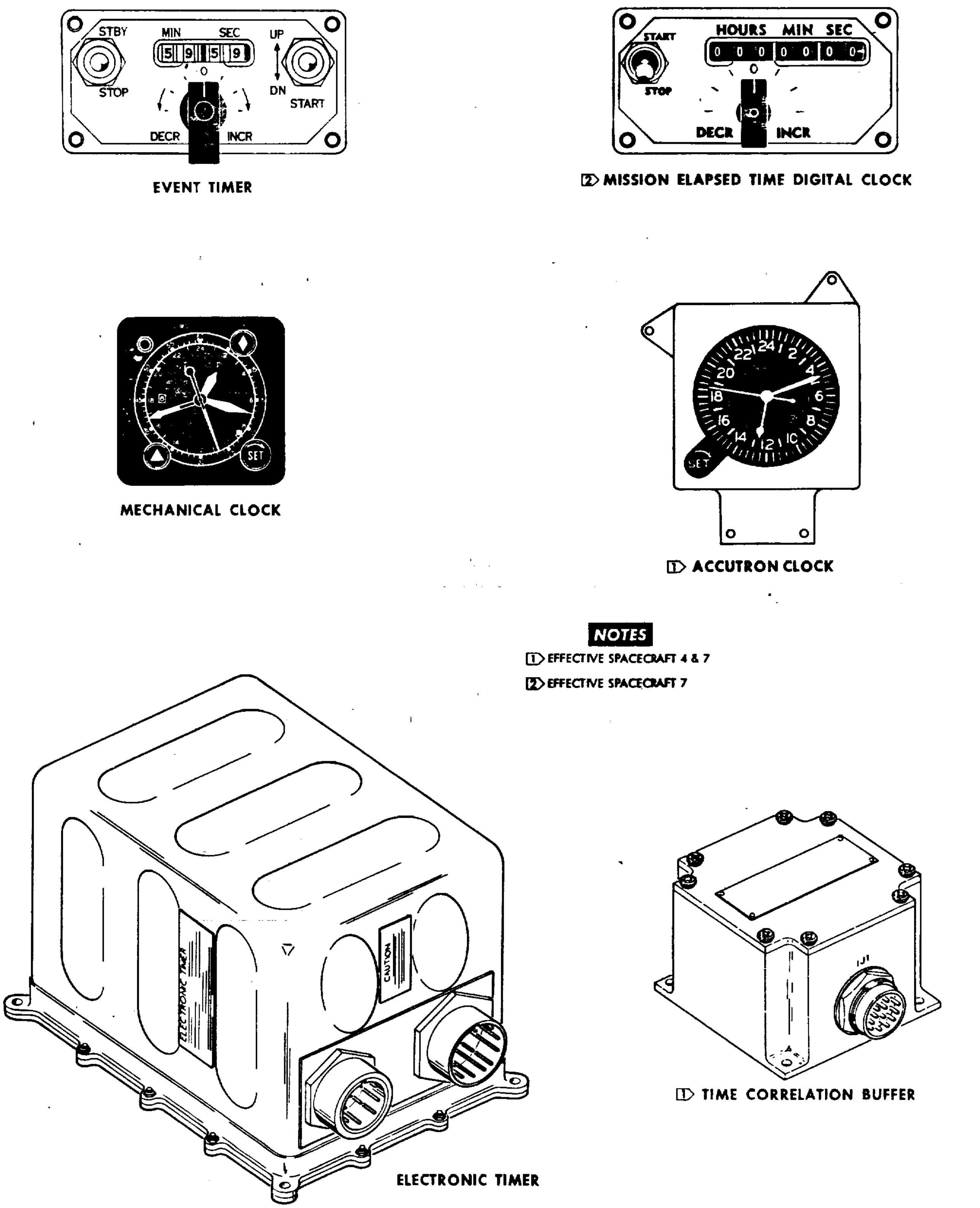
Construction

The electronic timer (Figure 8-62) is approximately 6" x 8 3/4" x 5 1/2" and weighs about ten pounds. It has two external connectors for interface with its associated systems. The enclosure for the unit is sealed to keep out moisture but is not pressurized. The timer utilizes a modular construction, containing eight modules which are wired directly into the enclosure. The modules are: (1) crystal oscillator, (2) timing assembly, (3) register control assembly, (4) memory control assembly, (5) memory assembly, (6) driver assembly, (7) relay assembly, and (8) power supply. Printed circuit boards and solid state components are used in all modules except the crystal oscillator.

Operation

The electronic timer is basically an electronic binary counter. It performs the counting operation for each of its functions (ET, TTG to T_R, and TTG to T_X) by an add/subtract program which is repeated every 1/8 second. (Refer to Figure 8-63). In each repetition of the counting operation, a binary word, representing ET or a TTG, is modified to represent a new amount of time. Magnetic core storage registers are used to store or remember the binary words between counting cycles. A storage register is provided for each of the three timer functions and another is provided for use as a buffer register for data transfer between the timer and the digital computer.

A crystal controlled oscillator is used as a frequency standard for developing the timing pulses necessary for the operation of the timer. This type of oscillator provides the high degree of accuracy required for the timer whose



NOTES

1 EFFECTIVE SPACECRAFT 4 & 7

2 EFFECTIVE SPACECRAFT 7

Figure 8-62 Time Reference System Components

FM1-8-62B

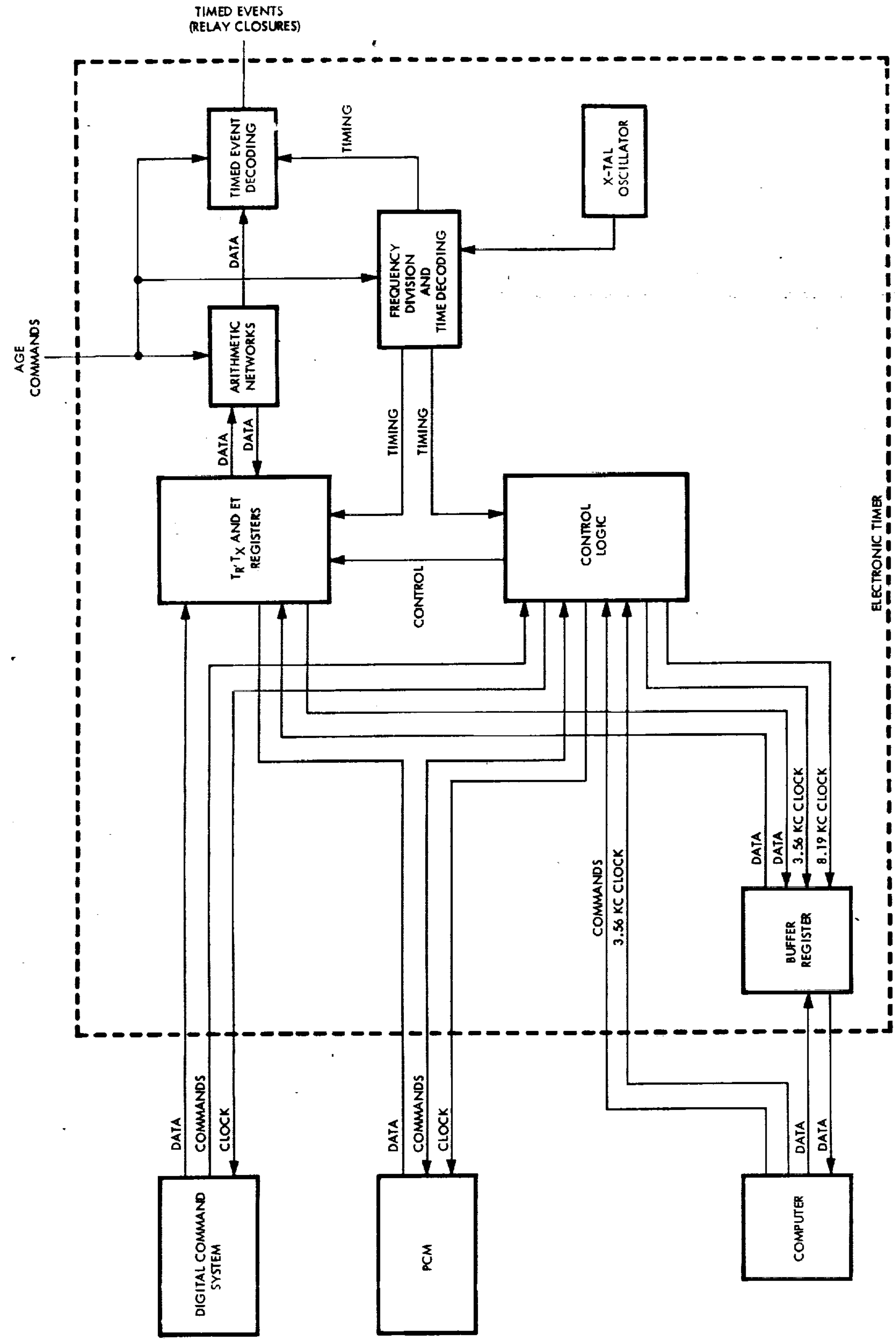


Figure 8-63 Electronic Timer Functional Block Diagram

FMG2-137

operations take place in very small fractions of a second. The oscillator is coupled to a series of toggle flip flops whose outputs provide the actual timing pulses for the timer operation.

The electronic timer utilizes a 32-word time program. That is, each 1/8 second of time is further divided into 32-word times. Each word time is divided into 32 bit times, and each bit time is divided into 32 "S" pulse times. "S" pulses are the shortest pulses used in the timer operation and are 3.8 microseconds long. One bit time is equal to 122 microseconds and one word time 3.9 milliseconds. It is pulses of these durations, and their multiples, which are produced by the toggle flip flops in the timing module.

Timer Start Circuit

Timer operation is initiated when a 28 VDC start signal is received from either the spacecraft Sequential System or the event timer. The signal from the Sequential System is transmitted to the electronic timer, automatically, at lift-off; the one from the event timer is generated when the UP/DN toggle switch on the face of the unit is placed in the UP position. Receipt of a signal from either source causes the set side of the clock-start relay to be actuated. Until lift-off, the relay is held in the reset position by a clock-hold signal from the AGE via the spacecraft umbilical. This is done to assure that the timer will not be started prematurely and will be at zero at the time of lift-off. Actuation of the clock-start relay causes a positive control signal to be applied to a gate in the timing module. This gate allows the output of the crystal controlled oscillator to be coupled to the countdown flip flops.

Countdown and Time Decoding

The countdown and time decoding operations take place primarily in the timing module. When timer operation is initiated, the 1.048576 megacycle output of the crystal-controlled oscillator is coupled to the first of a series of 17 toggle flip flops (refer to Figure 8-64). Twelve of the flip flops are contained in the timing module and five in the register control module. The flip flops form a frequency dividing network, each stage of which produces one square wave output pulse for every two input pulses. The output frequency of the final stage in the series is eight pulses per second.

Outputs of all but the first two stages of the countdown circuitry are utilized to develop the timing pulses necessary for timer operations. Output pulses from either the "1" or the "0" side of an individual flip flop may be used; however, the polarity of the pulses from one side will be 180° out of phase with those from the other side. Pulses from the flip flop outputs are supplied, in certain combinations, to gate circuits in the time decoding section. Each gate circuit receives several input pulse trains and produce output pulses which are usable for the timer circuitry (refer to Figure 8-65a). Basically, a gate will produce output pulses which will have the pulse width of the narrowest input pulses and the frequency of the input pulse train with the widest pulses. If the polarity of one input is reversed, the time at which the output pulse occurs, will change (refer to Figure 8-65b).

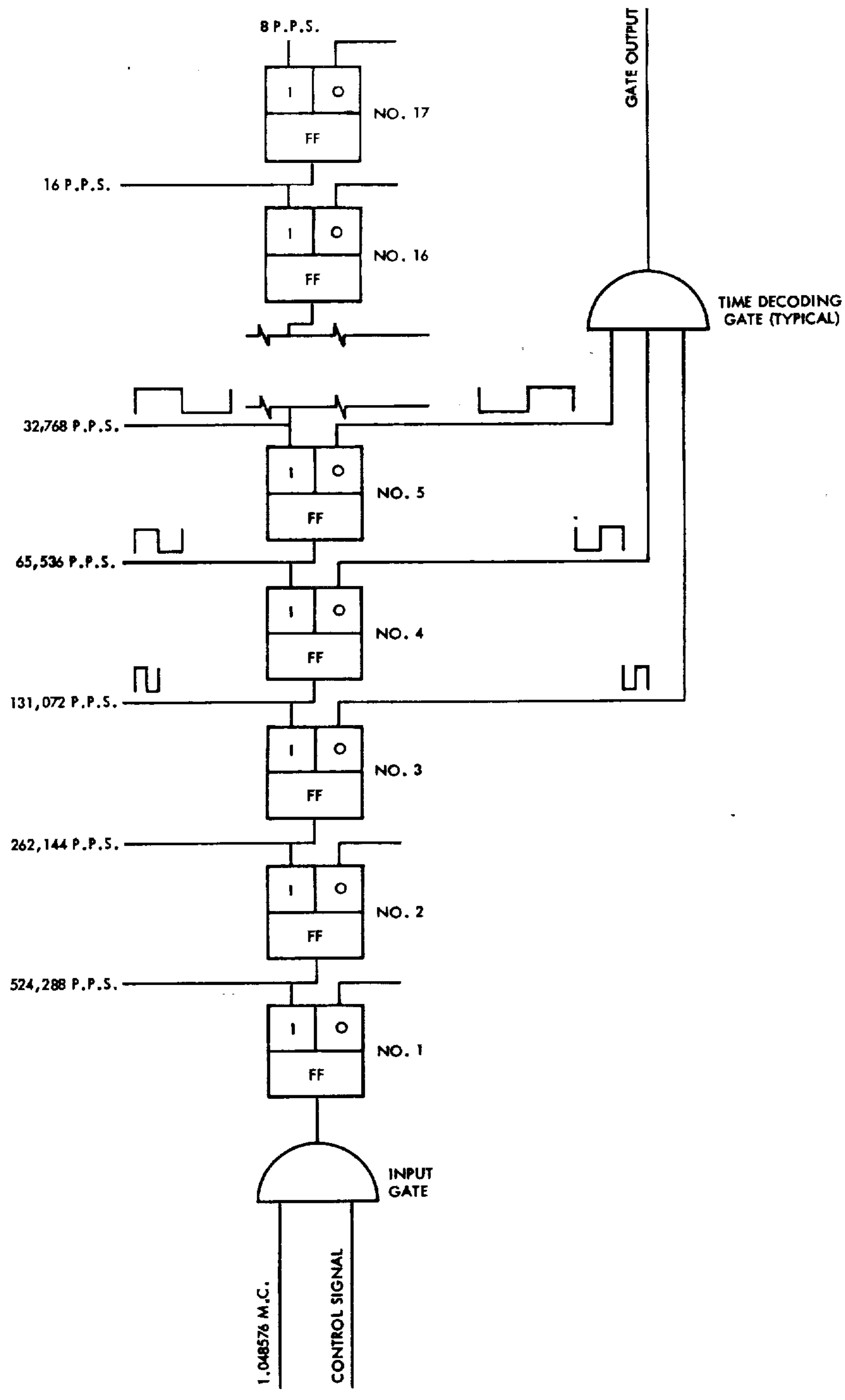


Figure 8-64 Schematic Diagram, Frequency Division & Time Decoding

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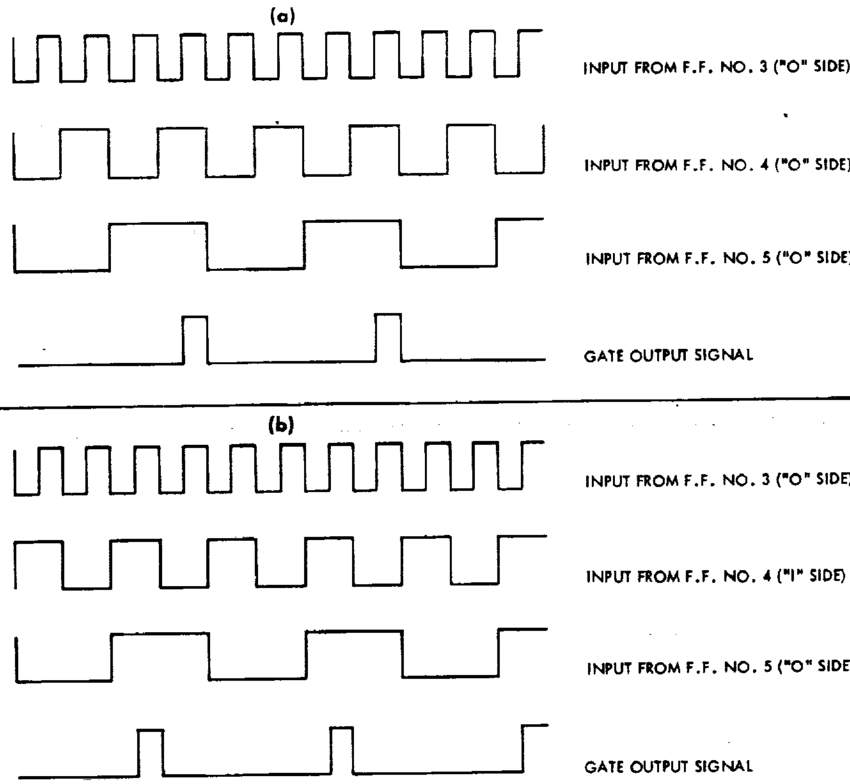

Figure 8-65 Time Decoding Gate Inputs and Outputs (Typical)

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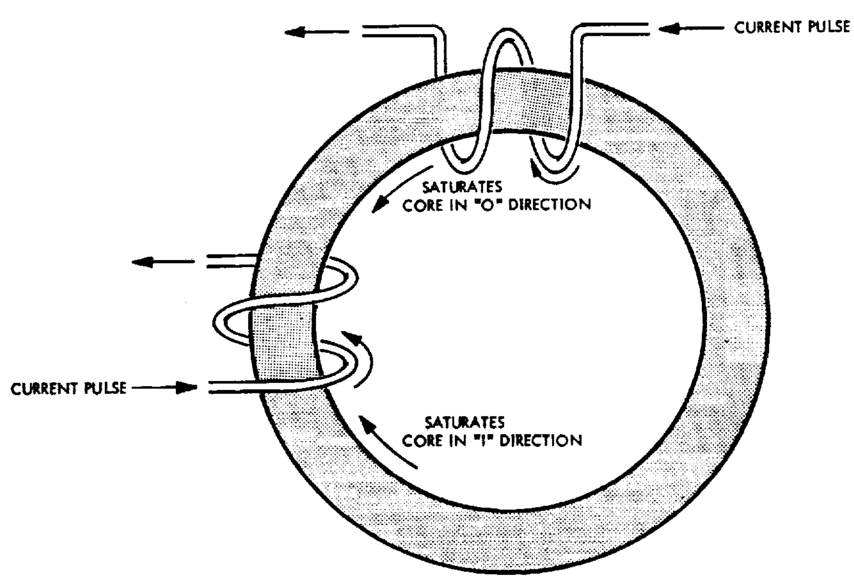


Figure 8-66 Magnetic Core Operation

FMG2-133

Operational Control

Two complete modules are required to encompass all of the circuitry necessary to perform the control functions in the electronic timer. The register control module primarily controls the transfer of data into and out of the timer. The memory control module directly controls the operations of the magnetic storage registers in the memory module.

The register control module supplies the control signals which are required to perform the operations directly associated with the transfer of time data. It utilizes the various command and clock signals from the other spacecraft systems to produce its control signals. The control signals are then supplied to the appropriate circuitry to: (1) receive a new binary data word (as in the updating process), (2) initiate the shifting operations of the proper storage registers to "write" in or "read" out the desired time data (ET, T_R, or T_X), and (3) supply data, read out of the storage registers, to the proper timer output terminal(s) to be transferred to the system requesting it.

The memory control module directly controls the operation of the magnetic storage registers and performs the arithmetic computations of the counting process. Inputs from the timing and register control modules are utilized to develop the shift and transfer output pulses for shifting data words into and out of the storage registers. These pulses are developed separately for each register.

Both control modules are made up of rather complex and overlapping networks of logic circuitry. The memory control module also employs shift current generators and transfer switches, as output stages, to develop the required power capabilities.

Storage Register Operation

The magnetic storage registers for ET, T_R, and T_X are used to store or remember binary words of time data. These data words may be shifted out of their respective registers, as required, for the counting operations and for transfer to other spacecraft systems. The transfer of data into and out of a storage register is accomplished, serially, with the Least Significant Bit (LSB) first.

A storage register is comprised of a series of magnetic memory cores, each of which is capable of storing one binary bit of time data. This capability is based upon the characteristic of a magnetic core to saturate in one of two directions when a current pulse is applied to one of its windings. (Refer to Figure 8-66.) Saturation in one direction represents a binary "1" and indicates the presence of a data bit. Saturation in the other direction represents a binary "0" and indicates the absence of a data bit. The storage registers for ET and TTG to T_R each contain 24 magnetic cores and the register for TTG to T_X contains 16. Therefore, a binary word for ET or T_R consists of 24 bits, while a word for T_X consists of 16 bits.

The use of the binary system for time representation permits the storage of data which can represent an amount of time as small as 1/8 second and as large as 24 days. Each data bit in a binary data word represents one individual increment of time. In looking at the flow diagram in Figure 8-67a the 24 sections of the storage register represent its 24 individual cores. The data bit which represents the smallest time increment (1/8 second) is stored in core number 24. It is referred to as the LSB in the data word. Core number 23, then, would store the next bit (representing 1/4 of a second) of the data word. The sequence continues, with core number 22 representing 1/2 second, back through core number 1 with each successive core representing a time increment twice that of the preceding one. By adding together the increments of time represented by all of the cores, the total time capacity of the register can be determined. Thus, it is found that the ET and T_R registers have capacities of approximately 24 days and the T_X register, approximately two hours. Conversion of a data word to its representative time may be accomplished by totaling the increments of time represented by the bit positions of the word where binary ones are present. For the data word shown in Figure 8-67b the representative time is 583 3/8 seconds.

The process of shifting a data word into or out of a storage register is controlled by the occurrence of the shift and transfer pulses and by the condition of a control gate preceding each register and its write-in amplifier. The shift and transfer pulses from the control section are supplied to a storage register whenever a data word is to be written in or read out. These pulses occur once each bit time for a duration of one word time. The actual flow of data into a storage register is controlled by a logic gate preceding the write-in amplifier for each register. (Refer to Figure 8-68.) The count enable input of the gate will have a continuously positive voltage applied after lift-off has occurred. The write-in pulse input will have a positive pulse applied for

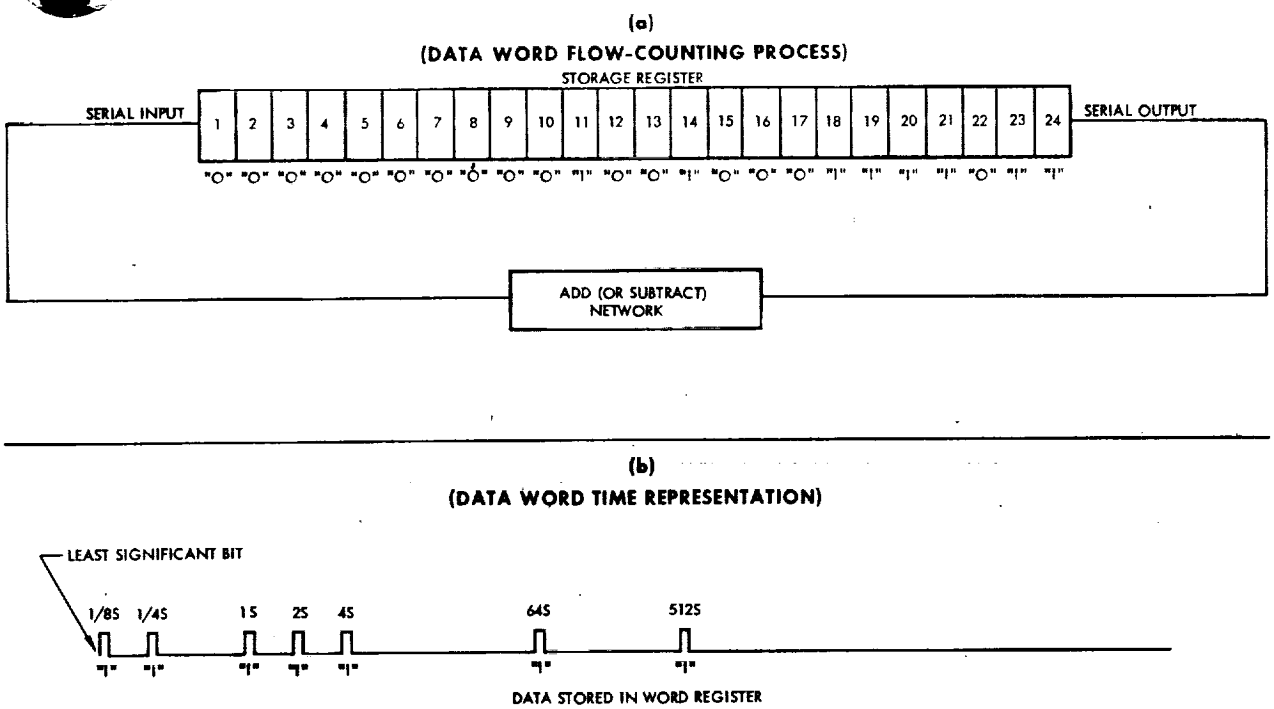


Figure 8-67 Time Data Word Flow & Representation

FMG2-134

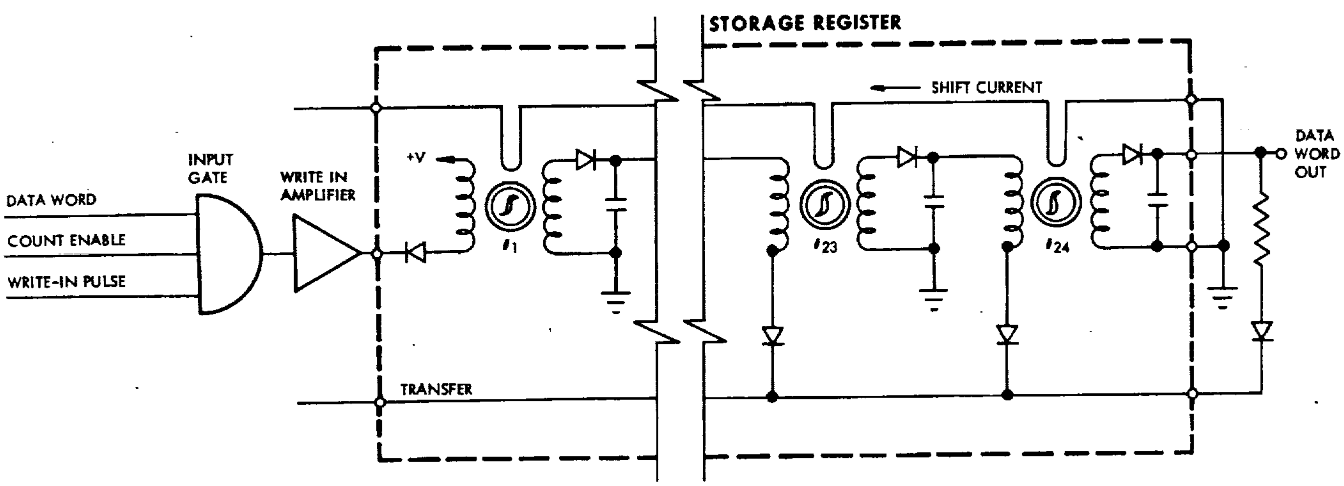


Figure 8-68 Schematic Diagram-Storage Register

FMG2-132

7.6 microseconds during each bit time (122 microseconds). These two inputs control the gate. The result is that a positive data pulse may pass through the gate only during a 7.6 microsecond period during each bit time.

When a binary data word is to be written into a storage register, its individual bits appear at the input of core number 1 as a series of current pulses. When the first current pulse (representing 1/8 second) of the word flows through the input winding of core number 1, the core is saturated in the binary "1" direction. It remains in this condition until a current pulse flows through the shift winding of the core. The shift pulse causes the flux of the core to collapse and reform, switching the core back to the "0" condition. When this occurs, a voltage is developed across the output winding of the core and the temporary storage capacitor is charged through the winding from the diode end. When the shift pulse decays and a ground potential is placed on the transfer line, the capacitor discharges through the input winding of the next core, setting it to the binary "1" condition. Whenever a bit position of the incoming data word does not contain a pulse, core number 1 is not switched to "1." As a result, its shift pulse causes no change of flux; no voltage is developed across the output and the capacitor is not charged or discharged. Hence the next core is not set to the "1" condition. Because the shift pulses are applied to all the cores in a register, simultaneously, it is assured that each one is set to the "0" condition before the transfer pulse (also applied to all cores, simultaneously) allows the storage capacitors to discharge. When a complete word has been written into the register, the cores which are in the binary "1" condition contain the binary data bits.

Reading a data word out of a storage register involves basically the same processes as writing one in.

The data bits shift from left to right, with the bit in core number 24 leaving the register first. An additional bit is shifted out of the register with each repetition of the shifting process.

Counting Operations

The counting operation for each of the timer functions consists of reading a binary data word out of a storage register, cycling it through an arithmetic network, and writing it back into the register. (Refer to Figure 8-65a.) The operation is completed in one word time and is repeated every 1/8 second. In the process, the time representation of the word is changed by increment of 1/8 second.

The read and write portions of the counting operation take place concurrently. As the first data bit is shifted out of a register, the remaining bits shift one core to the right, leaving core number 1 vacant. Before the next shift operation takes place, the bit which has been shifted out of the register is cycled, instantaneously, through the arithmetic circuitry and inserted back into core number 1. The process is the same for each bit of the word. Thus, when the last bit of the original word is shifted out of the register, the first bit of the new one shifts into core number 24. The last bit then cycles through the arithmetic circuitry and enters core number 1, completing the counting operation.

In the arithmetic portion of the counting process, the output of the elapsed time register is supplied to an add circuit and those from the T_R and T_X registers to separate subtract circuits. Both types of circuits are made up of combinations of logic and switching circuits. Their operation is quite similar, the main difference being in their logic programs.

The add process for the ET function consists of adding a binary "1" to the first bit position (the LSB) of the word coming into the add circuit. If there is already a "1" in that bit position, the "1" is carried to the next bit position. The carry operation continues until the "1" reaches an open bit position.

When the first bit of a data word read out of the ET register is a binary "0", the add circuit produces a positive output signal. The positive signal is then inverted by the write-in amplifier and supplied to the input of the storage register. With a negative input to the register, a binary "1" is written into core number 1 as the first bit of the new word. Thus, the first bit of the word has been changed from a binary "0" to a binary "1" adding 1/8 second to the representative time of the word. The remaining bits are written back into the register just as they were read out.

When a binary "1" is read out of the ET register as the first bit of a data word, the output of the add circuit will be negative. Upon inversion by the write-in amplifier, the signal will be positive. A positive signal at the register input causes a binary "0" to be written into the first core. If the subsequent, consecutive, data bits are also binary "1"'s, the output of the add circuit remains negative, causing binary "1"'s to be written into the register.

Upon receipt of the first binary "0" in the data word from the register, the output of the add circuit becomes positive, causing a binary "1" to be written back into the register for that bit position. For example, if the first five bits of the word being read out of the register are binary "1"'s (representing a total of 3 7/8 seconds of ET) and the next one is a binary "0", then the first five bits of the new word will be binary "0"'s; and the sixth will be a binary "1." A binary "1" in the sixth bit position represents an ET of four seconds. The remaining bits of the data word, again, are inserted back into the register just as they were read out.

Although the circuitry of a subtract network is much the same as that of an add network, the operation is different because of the subtract logic. If the LSB of a word coming into a subtract network is a binary "1", the output for that bit position will be negative, causing a binary "0" to be written back into register. In this case, the 1/8 second has now been subtracted, and the balance of the word will remain the same. If the LSB of the incoming word is a binary "0" the output of the subtract network will become positive, allowing a binary "1" to be written into the register. The output of the subtract circuitry will remain positive until the first binary "1" enters the circuitry. When this occurs, the output becomes negative and causes a binary "0" to be written into the register. The rest of the word is then written back into the register just as it came out.

Data Transfer

Binary words of time data are transferred into and out of the electronic timer by several different methods. Data words received from the ground station, via the Digital Command System, are inserted directly into their respective storage registers in the timer. Data from the guidance system computer, however, is transferred into the buffer register of the timer and then shifted into the proper storage register. The same process is involved in the transfer of data from the timer to the computer: a word is shifted out of its storage register into the buffer register and then transferred to the computer. Data transfer from the timer to the Instrumentation System is accomplished by shifting the desired data out of its register to a pulse transformer. The output of the transformer is coupled to a storage register in the Instrumentation System.

Timer Interfaces

The following is a list of the inputs and outputs of the electronic timer together with a brief description of each:

INPUTS

(a) A continuous 28 VDC signal from the spacecraft Sequential System at lift-off to start the recording of ET and countdown of T_R and T_X.

(b) A 28 volt emergency start signal from the event timer to initiate the electronic timer operation in the event that the lift-off signal is not received from the Sequential System. The signal would be crew-ground co-ordinated and would be initiated by actuation of the event timer UP-DN switch to UP.

(c) A read/write command signal from the digital computer to direct the timer as to which function is to be accomplished.

(d) A TTG to T_R address signal from the digital computer to update or readout TTG to T_R.

(e) A TTG to T_X address signal from the digital computer to enter a TTG to T_X.

(f) An elapsed time address signal from the digital computer to readout ET.

(g) Twenty-four clock pulses from the digital computer to accomplish data transfer. (25 pulses for data transfer out of the electronic timer)

(h) "Write" data for update of TTG to T_R, or TTG to T_X from the digital computer. Twenty-four data bits will be forwarded serially, LSB first.

(i) A TTG to T_R ready signal from the DCS to command update of TTG to T_R.

(j) A TTG to T_X ready signal from the DCS to command entry of a TTG to T_X.

(k) Serial data from the DCS to update TTG to T_R, or TTG to T_X. Twenty-four data bits will be forwarded serially, least significant bit first. Clocking is provided by the electronic timer.

(l) TTG to T_R readout signals from the Instrumentation System.

(m) An elapsed time readout signal from the Instrumentation System.

(n) An AGE/count inhibit signal from ground based equipment, via the spacecraft umbilical, to keep the elapsed time register at zero time prior to launch.

(o) A clock hold signal from ground based equipment, via the spacecraft umbilical, tc prevent the timer from operating prior to launch.

(p) An event relay reset signal from ground based equipment via the spacecraft umbilical.

(q) An event relay check signal from ground based equipment via the spacecraft umbilical.

OUTPUTS

(a) A contact closure at T_R for the digital computer.

(b) A contact closure at T_R (Continuous) for the Sequential System.

(c) A contact closure at T_X for the DCS.

(d) "Read" data to the digital computer for ET or TTG to T_R. Data bits are forwarded serially, LSB first.

(e) Signal power ($12 \,^{+0}_{-1}$ volts) to the DCS and Instrumentation System.

(f) Twenty-four clock pulses to the DCS to accomplish data transfer.

(g) Twenty-four clock pulses to the Instrumentation System to accomplish data transfer.

(h) Serial data to the Instrumentation System for readout of ET or TTG to T_R. Data bits are forwarded serially, least significant bit first.

(i) A contact closure from T_R-256 seconds on S/C 7 and T_R-5 minutes on S/C 3 and 4 for the Sequential System.

(j) A contact closure from T_R-30 seconds for the Sequential System.

(k) An input power monitor signal to ground based equipment via the spacecraft umbilical.

TIME CORRELATION BUFFER

General

The Time Correlation Buffer (TCB), used on S/C 4 and 7, supplies the time correlation signals for the bio-medical and voice tape recorders. Serial data and data clock output from the electronic timer is applied to the TCB input. Serial data contains 24 elapsed time words, and extra elapsed time word and a time to go to retrograde word. The TCB selects the extra elapsed time word and modifies the word format to make it compatible with the tape recorder frequency responses. Information to the recorder is updated once every 2.4 seconds and has the same resolution (1/8 second) as the electronic timer.

Construction

The dimensions of the TCB (Figure 8-62) are 2.77" X 3.75" X 3.80" and the weight is approximately 3.0 pounds. The TCB contains magnetic shift registers, a 100 KC astable multivibrator, a power supply and logic circuitry. One 19 pin connector provides both input and output connections.

Operation

The operation of the TCB is dependent on signals from the Instrumentation System and the electronic timer. In response to request pulses from the Instrumentation System, the electronic timer provides elapsed time and time-to-go to retrograde words to both the instrumentation system and the TCB. The elapsed time word is supplied every 100 milliseconds. In addition, once every 2.4 seconds it provides an extra elapsed time word and 100 milliseconds later it provides a time to go to retrograde word.

The TCB requires elapsed time information only; therefore, the time to go to retrograde word is rejected. The tape recorders, due to their response times, are not capable of recording time data every 100 milliseconds and for this reason, only the extra elapsed time word is accepted by the TCB. The remaining 24 elapsed time words and the time to go to retrograde word are rejected by logic circuitry in the TCB. Rejection of unused words is based on their time relationship to other words.

The TCB contains three 8-bit magnetic shift registers in which the 24 bit extra elapsed time word is loaded once every 2.4 seconds. The TCB then shifts out bits at the rate of one every 100 milliseconds. The shift rate is based on data clock pulses from the electronic timer. The first data clock pulse in a word causes the TCB to shift out one bit of the data and the other 23 data clock pulses are disregarded.

Each bit that is shifted out of the shift register is stretched in time and coded to make it compatible with tape recorder response times. The output to the bio-medical recorder is one positive pulse for a binary "0" and two positive pulses for a binary "1." The most significant bit has two additional pulses to distinguish it from the other 23 bits in the word. Data is shifted out of the TCB in a least significant bit first and most significant or marker bit last.

The output to the voice tape recorder is the same basic format as for the bio-medical recorders. However, to make it compatible with the higher frequency response characteristics of the voice tape recorder, each output pulse is chopped into two pulses, doubling the frequency.

All input and output signals are coupled through isolation transformers providing complete DC isolation.

MISSION ELAPSED TIME DIGITAL CLOCK

The mission elapsed time digital clock (used on S/C 7) is capable of counting time up to a maximum of 999 hours, 59 minutes and 59 seconds. The time is displayed on a decimal display indicator on the face of the unit. The seconds tumbler of the display is further graduated in 0.2 second increments. Counting may be started or stopped manually or by a remote signal. Prior to initiating a counting operation, the indicator should be manually preset to the desired starting time.

Construction

The dimensions of the digital clock are approximately 2 inches by 4 inches by 6 inches and its weight is approximately 2 pounds. On the face of the clock there are two controls and a decimal display window. The unit contains four electronic modules, a relay and a step servo motor. A gear train connects the servo motor with the decimal display tumblers. An electrical connector is provided at the rear of the unit for power and signal inputs.

Operation

Operation of the digital clock is dependent on timing pulses from the electronic timer. The time base used for normal counting operations in the digital clock is derived from the 8 pps timing pulse output of the electronic timer. The 8 pps signal is buffered and used to establish the repetition rate of a step servo motor. The step servo motor is coupled through a gear train to display tumblers. Additional counting rates are selectable for the purpose of setting the clock to a desired starting point.

Start/Stop Operation

Remote starting of the digital clock is accomplished by providing the 8 pps timing pulses from the electronic timer. Before remote starting can be accomplished, the START/STOP switch must be in the START position and the DECR/INCR switch must be in the 0 position. Manual starting of the digital clock can be accomplished (if timing pulses are available) by placing the START/STOP switch in the START position. This energizes the start side of the start/stop relay. The relay applies control and operating voltages to the counting circuitry, allowing the counting operation to begin. Counting may be stopped by removing the time base (8 pps) from the clock or by placing the START/STOP switch in the STOP position, removing voltage and disabling the circuitry.

Counting Operations

When the start/stop relays are actuated and operating voltage of plus 28 volts DC applied to the servo motor, a plus 12 volt DC enable signal is applied to the normal count gate. This initiates the counting sequence. The electronic timer provides an 8 pps timing signal which is buffered and supplied to the sequential logic section.

Sequential logic section consists of four set-reset flip flops which provide the necessary sequences of output signals to cause the servomotor to step in one direction or the other (Figure 8-69). As the counting process begins, three of the flip flops are in the reset condition (reset output positive) and one is in the set condition (set output positive). With receipt of the first timing

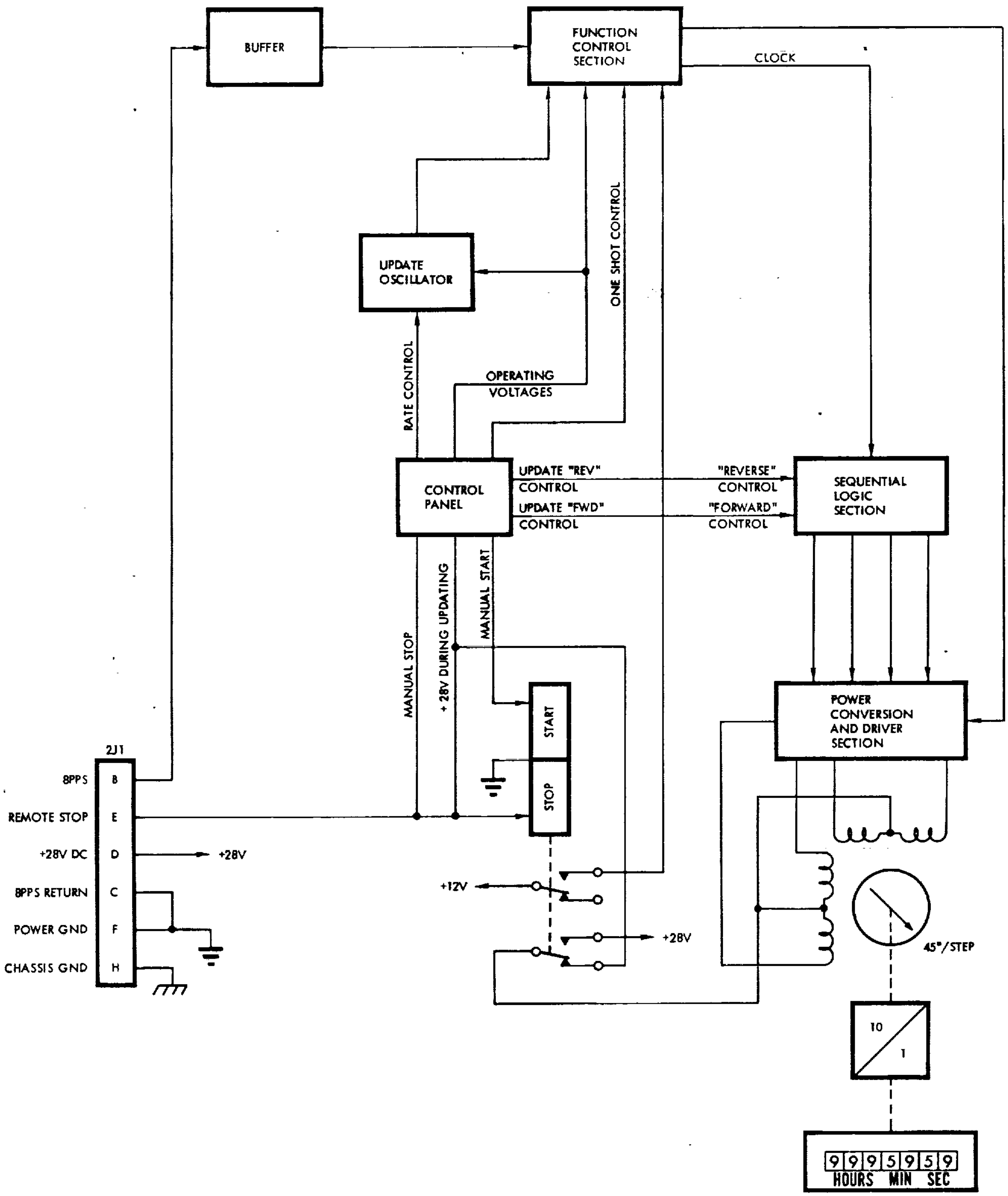


Figure 8-69 Mission Elapsed Time Digital Clock Functional Diagram

FM1-8-69

pulse, the next flip flop switches to the set condition. The first one also remains set, but the other two remain reset. Then, when another timing pulse is received, the first flip flop resets, leaving only the second one set. The sequence continues with alternate timing pulses setting one flip flop, then resetting the preceding one. After the fourth flip flop has been set and the third one subsequently reset, the first one is again switched to the set condition and the sequence is started over again. In order to have the logic section function properly, either a forward or reverse control signal must be received from the start/stop relay. These are used as steering signals for the timing pulses which set and reset the flip flops. For counting up, the control signals cause the flip flop operating sequence to be in one direction. When counting down, they cause the sequence to reverse: flip flop number 4 is set first, then number 3, etc., back through number 1. The output of the sequential logic circuit is applied to the power conversion and driver section.

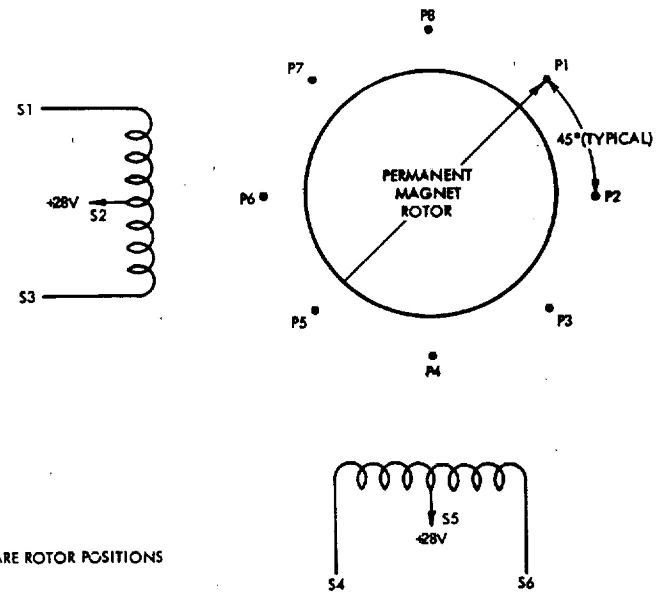
The power conversion and driver section converts the voltage-pulse outputs of the logic section to current pulses which are used to drive the servomotor. The driver section provides four separate channels, one for each input. Each channel has a logic gate and a power driver. The logic gate permits the logic section output to be sensed at ten selected times each second. The gate senses only the occurrence of a positive signal which will allow the power driver to conduct and send a pulse of current through one of the four servomotor stator windings.

The sequence of pulses from the driver section causes the servomotor to step eight times each second and 45° each step. Figure 8-70 illustrates the step positions relative to the sequence of operating pulses from the driver section. If pulses were applied to each of the four servomotor windings, without overlap, the unit would step 90° each repetition. It is this overlapping of signal applications which causes it to step 45° at a time.

The display indicator is a rotating counter with wheels to display seconds, tens of seconds, minutes, and tens of minutes, hours, tens of hours and hundreds of hours. It is coupled to the servomotor through a gear train with a reduction ratio, from the servomotor, of 10:1. Therefore, as the servomotor rotates 360° (in one second), the indicator shaft turns 36° or 1/8 of a rotation. Since the seconds wheel is directly coupled to the shaft and is calibrated from zero to nine, a new decimal is displayed each second. As the seconds wheel moves from nine to zero, the tens-of-seconds wheel moves to the one position. The operations of the other wheels are similar.

Updating

The display may be returned to zero or updated to some other readout with the use of the DECR-INCR rotary switch on the face of the timer. The rotary switch must be in the 0 position in order to have the timer operate at a normal rate; with the switch in one of the other positions, it counts at a different rate. There are three rate selections, each, for the INCR and DECR (count-up and count-down) updating modes. The positions on each side that are farthest from the 0 position are utilized to make the timer count at 25 times its normal rate.



NOTE
(1) P1 - P8 ARE ROTOR POSITIONS

OPERATION	RESULT
GROUND S1, GROUND S6	ROTOR INDEXES TO ARBITRARY REF. POSITION (P1)
OPEN S1	ROTOR STEPS 45° C.W. (P2)
GROUND S3	ROTOR STEPS 45° C.W. (P3)
OPEN S6	ROTOR STEPS 45° C.W. (P4)
GROUND S4	ROTOR STEPS 45° C.W. (P5)
OPEN S3	ROTOR STEPS 45° C.W. (P6)
GROUND S1	ROTOR STEPS 45° C.W. (P7)
OPEN S4	ROTOR STEPS 45° C.W. (P8)
GROUND S6	ROTOR RETURNS TO REF. POSITION (P1)
OPEN S6	ROTOR STEPS 45° C.C.W. (P8)
GROUND S4	ROTOR STEPS 45° C.C.W. (P7)
OPEN S1	ROTOR STEPS 45° C.C.W. (P6)
GROUND S3	ROTOR STEPS 45° C.C.W. (P5)
OPEN S4	ROTOR STEPS 45° C.C.W. (P4)
GROUND S6	ROTOR STEPS 45° C.C.W. (P3)
OPEN S3	ROTOR STEPS 45° C.C.W. (P2)
GROUND S1	ROTOR RETURNS TO REF. POSITION (P1)

Figure 8-70 Step Servomotor Operation

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The next closer positions are utilized to count at three times the normal rate. The positions nearest the 0 position are used to count at a rate 0.3 times the normal one. This position serves to more accurately place the indicator at a desired readout.

Operationally, positioning the rotary switch in some position other than 0 causes the time base frequency from the electronic timer to be replaced in the circuitry by an update oscillator. The frequency of the oscillator is established by the position of the rotary switch. In the 25X positions, the frequency is 400 cycles per second; in the 3X position, it is 48 cps; and in the 0.3X positions, it is approximately 4.8 cycles per second. The accuracy of the oscillator output is not critical since the oscillator functions only for updating purposes.

EVENT TIMER

General

The event timer is capable of counting time, either up or down, to a maximum of 99 minutes and 59 seconds on S/C 3 and 4 and to 59 minutes and 59 seconds on S/C 7. The timer is capable of counting time down to zero from any preselected time, up to the maximum listed above.

NOTE

> When the event timer is counting down, it will continue through zero if not manually stopped. After counting through zero, the timer will begin counting down from 99 minutes and 59 seconds on S/C 3 and 4 or 59 minutes and 59 seconds on S/C 7.

The time is displayed on a decimal display indicator on the face of the unit. The seconds tumbler of the display indicator is further graduated in 0.2 second increments. Counting, in either direction, may be started or stopped either remotely or manually. Prior to starting a counting operation, the indicator must be manually preset to the time from which it is desired to start counting.

Construction

The dimensions of the event timer are approximately 2" x 4" x 6" and the weight about two pounds. On the face of the timer, there are two toggle switches, one rotary switch, and a decimal display window. (Refer to Figure 8-62.) In addition to the panel-mounted controls, the unit contains four electronic modules, two relays, a tuning fork resonator, and a step servomotor. A gear train connects the servomotor with the decimal display tumblers. There is one electrical connector on the back of the unit.

Operation

The operation of the event timer is independent of the electronic timer. (Refer to Figure 8-71.) It provides its own time base which is used to control the operation of the decimal display mechanism. The time base used for normal counting operation is developed when the output of a tuning fork resonator is connected to a series of toggle-type flip flops. The resulting signal established the repetition rate of a step-type servomotor. The servomotor is coupled, through a gear train, to the display tumblers. Additional counting rates may be selected in order to rapidly reset the timer to zero or to some other desired indication.

Start/Stop Operations

The remote and manual start/stop functions of the timer are accomplished in almost exactly the same manner. The difference is only in the source of the control signals. In order to initiate counting operations by either method, it is necessary to first have the STOP-STBY toggle switch in either the STBY or the center off position. (Refer to Figure 8-62.)

NOTE

When starting is accomplished with the STOP-STBY switch in the center position, a small inaccuracy is incurred. To prevent any starting inaccuracies, place the STOP-STBY switch in the STBY position before starting the timer.

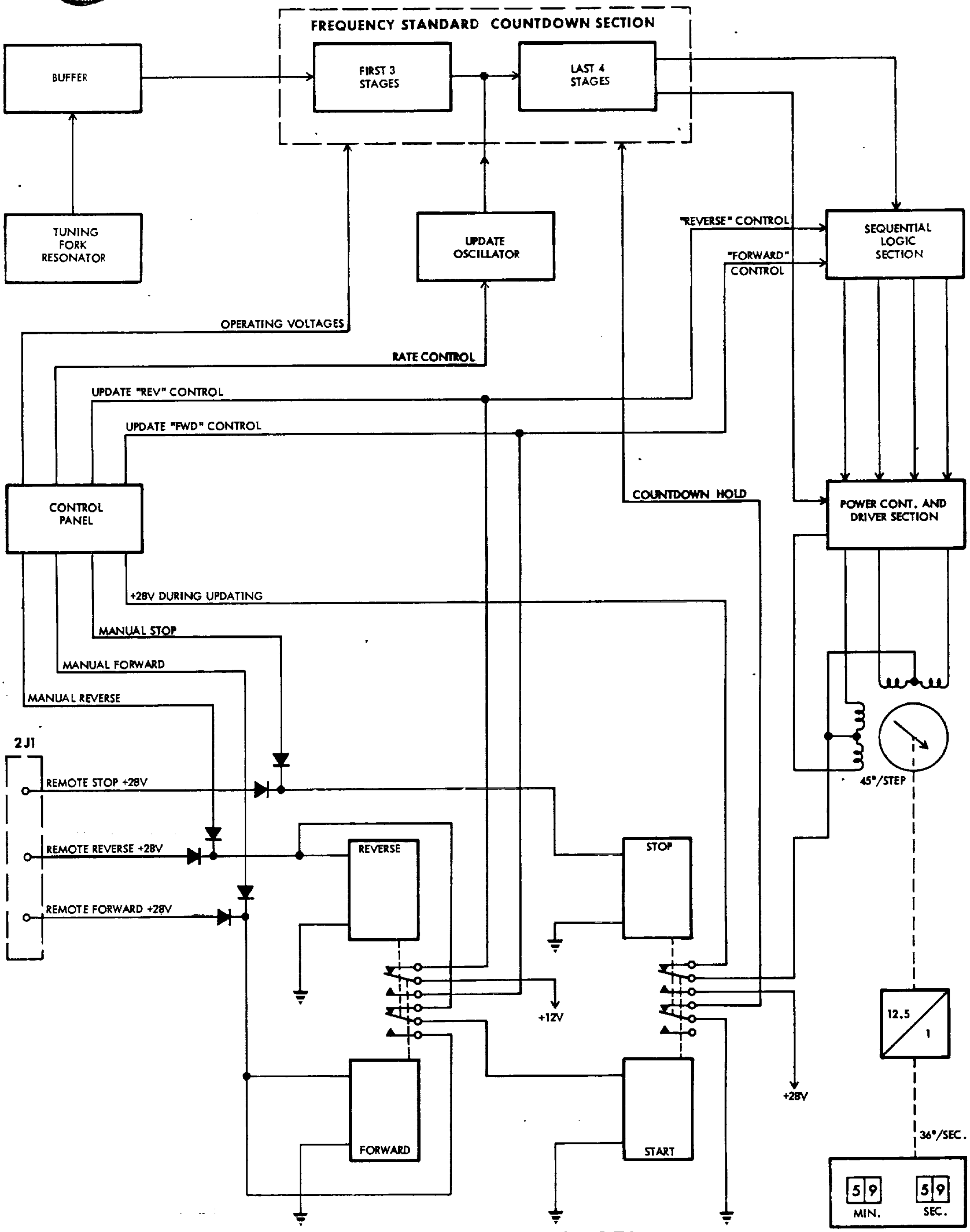

Figure 8-71 Event Timer Functional Diagram

FM1-8-69A

Manual starting may then be accomplished by placing the UP-DN toggle switch in either the UP or the DN position. This energizes one of the two coils of the forward/reverse relay, also causing the start coil of the start/stop relay to be energized. When these events take place, control and operating voltages are supplied to the counting circuitry, thus allowing the operation to begin. When starting is to be accomplished remotely, either a remote forward or a remote reverse signal is transmitted from the ground station to energize the forward/reverse relay. The counting process may be stopped upon receipt of a remote stop signal or by placing the STOP-STBY switch in the STOP position. Either of these functions energizes the stop side of the start/stop relay, removing critical operating voltages from the counting circuitry.

Counting Operations

Normal counting operations begin with the actuation of the forward/reverse relay in either direction and the start/stop relay in the start direction. When the forward/reverse and the start/stop relays are actuated, an operating voltage of +28 VDC is applied to the servomotor and a ground level inhibit signal is removed from the toggle flip flops. Also, a +12 VDC control signal, denoting either a forward or reverse counting process, is transmitted to the logic circuitry preceding the servomotor. The remainder of the timer circuitry has operating voltages applied when the STOP/STBY switch is placed in STBY.

With the application of operating voltages, the tuning fork resonator emits an AC signal of 1280 cycles per second. The signal is passed through a buffer to condition it for use by the series of seven toggle flip flops in the frequency standard countdown section. Since the output frequency of each flip flop is half that of its input, the final one in the series generates a signal of ten pulses per second. The outputs of the countdown section are connected to the sequential logic section and the power conversion and driver section.

Sequential logic section consists of four set-reset flip flops which provide the necessary sequences of output signals to cause the servomotor to step in one direction or the other (Figure 8-71). As the counting process begins, three of the flip flops are in the reset condition (reset output positive) and one is in the set condition (set output positive). With receipt of the first timing pulse, the next flip flop switches to the set condition. The first one also remains set, but the other two remain reset. Then, when another timing pulse is received, the first flip flop resets, leaving only the second one set. The sequence continues with alternate timing pulses setting one flip flop, then resetting the preceding one. After the fourth flip flop has been set and the third one subsequently reset, the first one is again switched to the set condition and the sequence is started over again. In order to have the logic section function properly, either a forward or reverse control signal must be received from the forward/reverse relay. These are used as steering signals for the timing pulses which set and reset the flip flops. For counting up, the control signals cause the flip flop operating sequence to be in one direction. When counting down, they cause the sequence to reverse: flip flop number 4 is set first, then number 3, etc., back through number 1.

The power conversion and driver section converts the voltage-pulse outputs of the logic section to current pulses which are used to drive the servomotor. The driver section provides four separate channels, one for each input. Each channel has a logic gate and a power driver. The logic gate permits the logic section output to be sensed at ten selected times each second. The gate senses only the occurrence of a positive signal which will allow the power driver to conduct and send a pulse of current through one of the four servomotor stator windings.

The sequence of pulses from the driver section causes the servomotor to step ten times each second and 45° each step. Figure 8-69 illustrates the step positions relative to the sequence of operating pulses from the driver section. If pulses were applied to each of the four servomotor windings, without overlap, the unit would step 90° each repetition. It is this overlapping of signal applications which causes it to step 45° at a time.

The display indicator is a rotating counter with wheels to display seconds, tens of seconds, minutes, and tens of minutes. It is coupled to the servomotor through a gear train with a reduction ratio, from the servomotor, of 12.5;1. Therefore, as the servomotor rotates 450° (in one second), the indicator shaft turns 36° or 1/10 of a rotation. Since the seconds wheel is directly coupled to the shaft and is calibrated from zero to nine, a new decimal is displayed each second. As the seconds wheel moves from nine to zero, the tens-of-seconds wheel moves to the one position. The operations of the other wheels are similar.

Updating

The display may be returned to zero or updated to some other readout with the use of the DECR-INCR rotary switch on the face of the timer. The rotary switch must be in the 0 position in order to have the timer operate at a normal rate;

with the switch in one of the other positions, it counts at a different rate. There are three rate selections, each, for the INCR and DECR (count-up and count-down) updating modes. The positions on each side that are farthest from the 0 position are utilized to make the timer count at 25 times its normal rate. The next closer positions are utilized to count at four times the normal rate. The positions nearest the 0 position are used to count at a rate 0.4 times the normal one. This position serves to more accurately place the indicator at a desired readout.

Operationally, positioning the rotary switch in some position other than 0 causes the tuning fork resonator and the first three toggle flip flops to be replaced in the circuitry by an update oscillator. The frequency of the oscillator is established by the position of the rotary switch. In the 25X positions, the frequency is 4,000 cycles per second; in the 4X position, it is 640 cps; and in the 0.4X positions, it is approximately 64 cycles per second. The accuracy of the oscillator output is not critical since the oscillator functions only for updating purposes.

ACCUTRON CLOCK

The Accutron clock (Figure 8-62), located on the command pilot's control panel, is used on S/C 4 and 7. The clock is approximately 2 3/8 inches square and one inch thick. The clock has a 24 hour dial with major divisions on the half hour. An hour hand, minute hand and a sweep second hand are provided for a precise indication of the time of day. The unit is completely self contained and has no electrical interface with the spacecraft. The clock is capable of operating continuously for approximately one year on the internal mercury battery.

Operation

The Accutron clock is provided with one control knob. The knob is used to stop, set and start the timer as desired. To stop the timer, the control is depressed. From the depressed position, the clock can be set to the desired time. The clock will start automatically when the control knob is released.

The Accutron clock is a highly accurate device with an error of less than $\pm$ 3 seconds per day. This high degree of accuracy is made possible by using a tuning fork as the time standard, instead of the conventional balance wheel and hair spring. The tuning fork is magnetically driven at a natural frequency of 360 cps. The tuning fork frequency is adjustable, making precise calibration of the clock possible. The vibrational motion of the tuning fork is converted to rotational motion by a jeweled pawl and ratchet system. The rotary motion is then appropriately geared to provide outputs of: one revolution per day, one revolution per hour and one revolution per minute, for the clock hands.

MECHANICAL CLOCK

Construction

The mechanical clock (Figure 8-62) is approximately 2 1/4" x 2 1/4" x 3 1/4" and weighs about one pound. The dial face is calibrated in increments of 0-24 and 0-60. The clock has two hands for the time of day portion and two for the stopwatch portion. The controls for operating both portions of the clock are located on the face of the unit.

Operation

The clock is a mechanical device which is self-powered and requires no outside inputs. The hand and dial-face clock displays Greenwich Mean Time (GMT) in hours and minutes. A control on the face provides for winding and setting the unit. With the passing of each 24-hour period, the calendar date indicator advances to the next consecutive number. The stopwatch portion of the clock can be started, stopped, and returned to zero at any time. Two settable markers are provided on the minute dial to provide a time memory, permitting the clock to serve as a short-term back-up timer.

PROPULSION SYSTEMS

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INOPERATIVE ON S/C #3

ORBIT ATTITUDE MANUEVERING SYSTEM	
5 6	PITCH UP
1 2	PITCH DOWN
3 4	YAW RIGHT
7 8	YAW LEFT
3 7	ROLL CLOCKWISE
4 8	ROLL COUNTERCLOCKWISE
9 10	TRANSLATE FORWARD
11 12	TRANSLATE AFT
13	TRANSLATE RIGHT
14	TRANSLATE LEFT
15	TRANSLATE UP
16	TRANSLATE DOWN

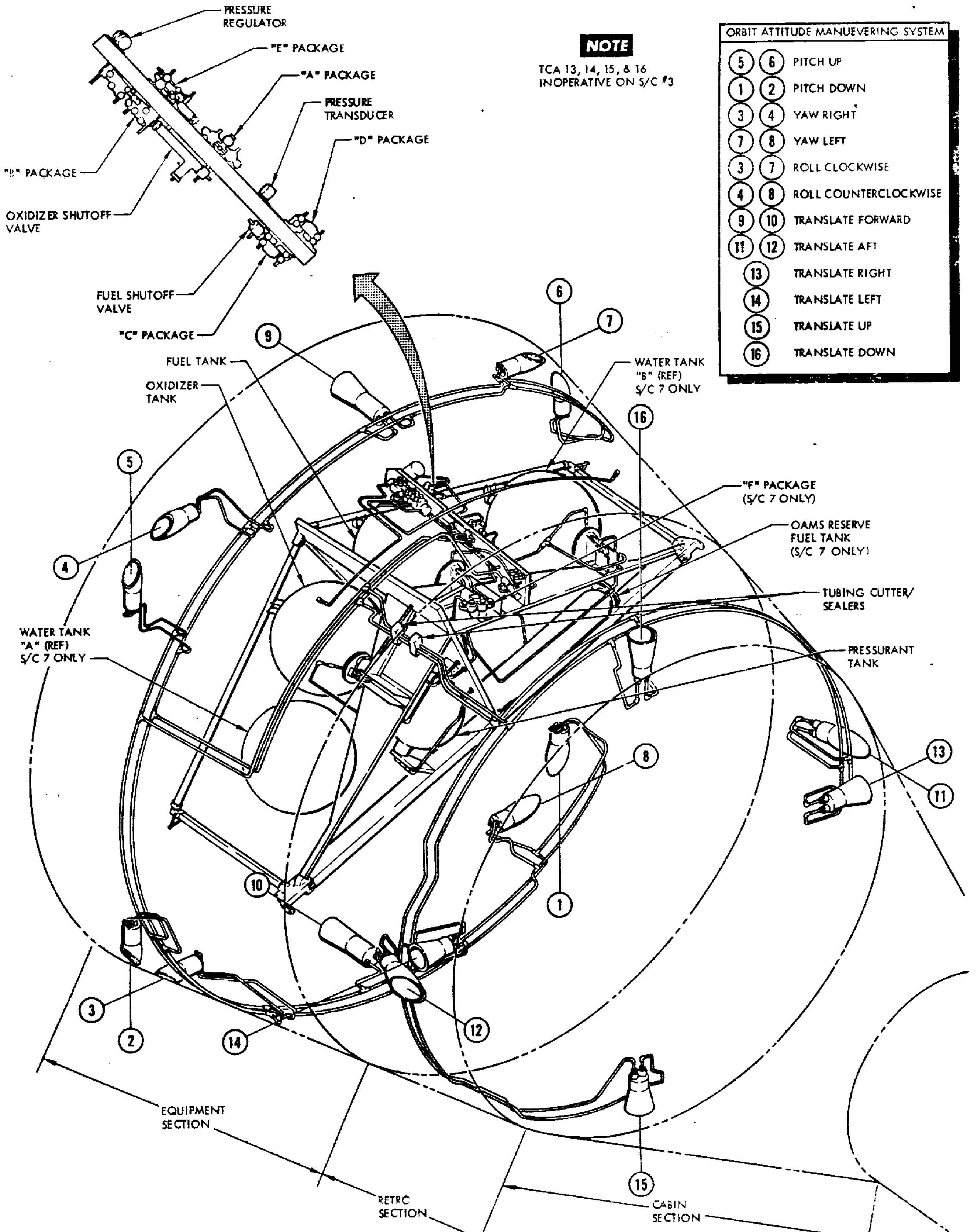

Figure 8-72 Orbit Attitude Maneuvering System and TCA Location

PROPULSION SYSTEM

GENERAL INFORMATION

The Gemini Spacecraft is provided with an attitude and maneuvering control capability. (Figure 8-72). This control capability is used during the entire spacecraft mission, from the time of launch vehicle separation until the re-entry phase is completed. Spacecraft control is accomplished by two rocket engine systems, the Orbit Attitude and Maneuvering System (OAMS) and the Re-entry Control System (RCS).

The OAMS controls the spacecraft attitude and provides maneuver capability from the time of launch vehicle separation until the initiation of the retrograde phase of the mission. The RCS provides attitude control for the re-entry module during the re-entry phase of the mission. The OAMS and RCS respond to electrical commands from the Attitude Control Maneuvering Electronics (ACME) in the automatic mode or from the crew in the manual mode.

ORBIT ATTITUDE AND MANEUVERING SYSTEM

SYSTEM DESCRIPTION

The Orbit Attitude Maneuvering System (OAMS) (Figure 8-72) is a fixed thrust, cold gas pressurized, storable liquid, hypergolic bi-propellant, self contained propulsion system, which is capable of operating in the environment outside the earth's atmosphere. Maneuvering capability is obtained by firing thrust chamber assemblies (TCA) singly or in groups. The thrust chamber assemblies are mounted at vaious points about the adapter in locations consistent with the modes of rotational or translation acceleration required.

The OAMS provides a means of rotating the spacecraft about its three attitude control axes (roll, pitch, and yaw) and translation control in six directions (right, left, up, down, forward and aft). The combination of attitude and translational maneuvering creates the capability of rendezvous and docking with another space vehicle in orbit. Spacecraft 3 does not have the capability to translate up, down, left or right.

The primary purpose of OAMS is spacecraft control in orbit. The OAMS is also used, after firing of shaped charges, to separate the spacecraft from the launch vehicle during a normal launch or in case of an abort which may occur late in the launch phase. During initiation of retrograde sequence, tubing cutter/sealer devices sever and seal the propellant feed lines from the equipment adapter. All of the OAMS (except six TCA's located in retro section) are separated from the spacecraft with the equipment section of the adapter. Spacecraft control functions are then assumed by the Re-entry Control System (RCS). OAMS control units and tanks are mounted on a structural frame (module concept) in the equipment section. The control units consist of forged and welded "packages". Each package consists of several functioning components and filters. The delivery of pressurant, fuel and oxidizer is accomplished by a uniquely brazed tubing manifold system. The OAMS system is divided into three groups; pressurant group, fuel/oxidizer group and thrust chamber assembly (TCA) group.

Pressurant Group

The pressurant group (Figure 8-73) consists of a pressurant tank, "A" package, "E" package, "F" package on Spacecraft 7, pressure regulator, and "B" package. Inlet valves, ports and test ports are provided at accessible points to permit servicing, venting, purging and testing. Filters are provided throughout the

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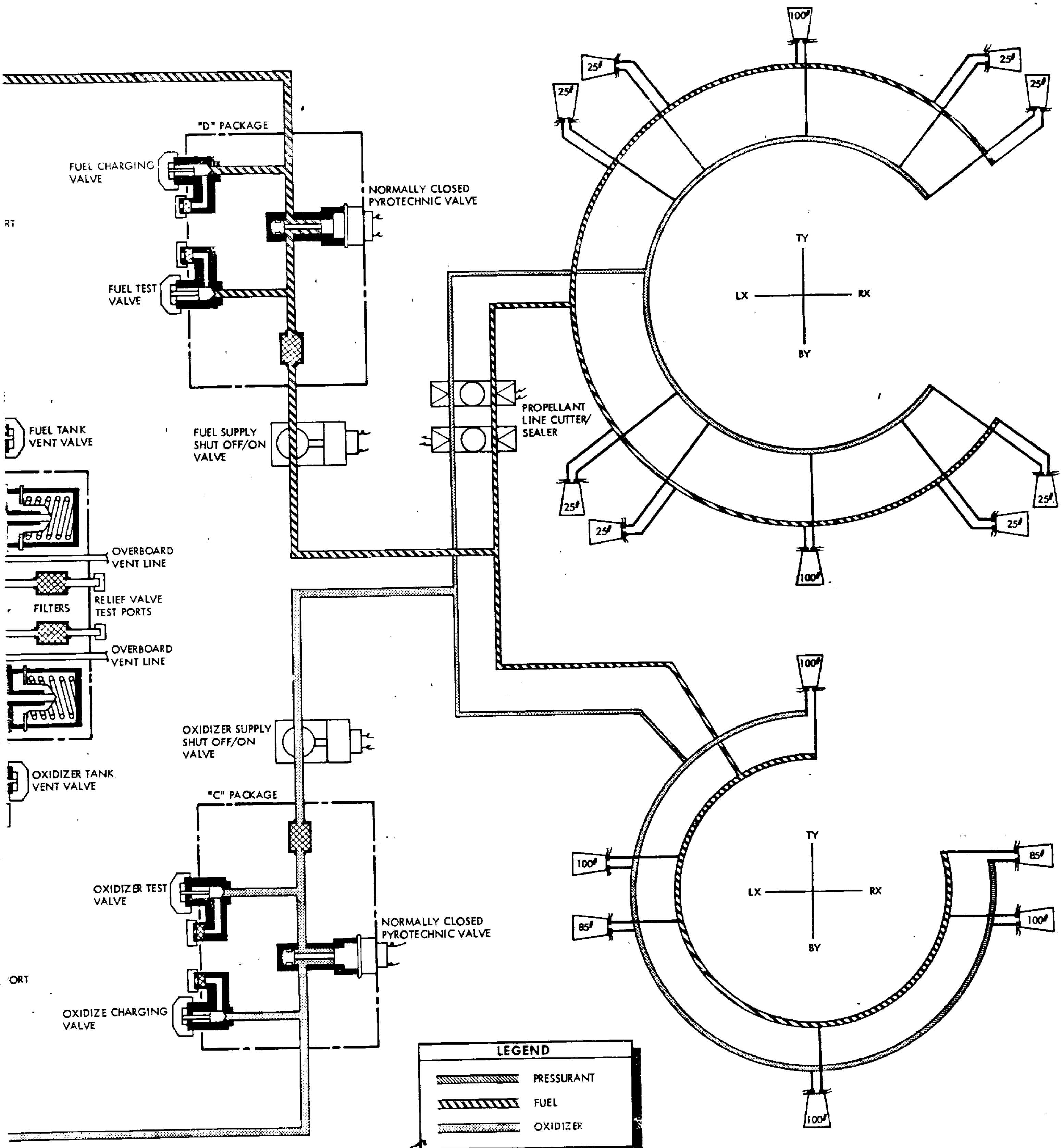

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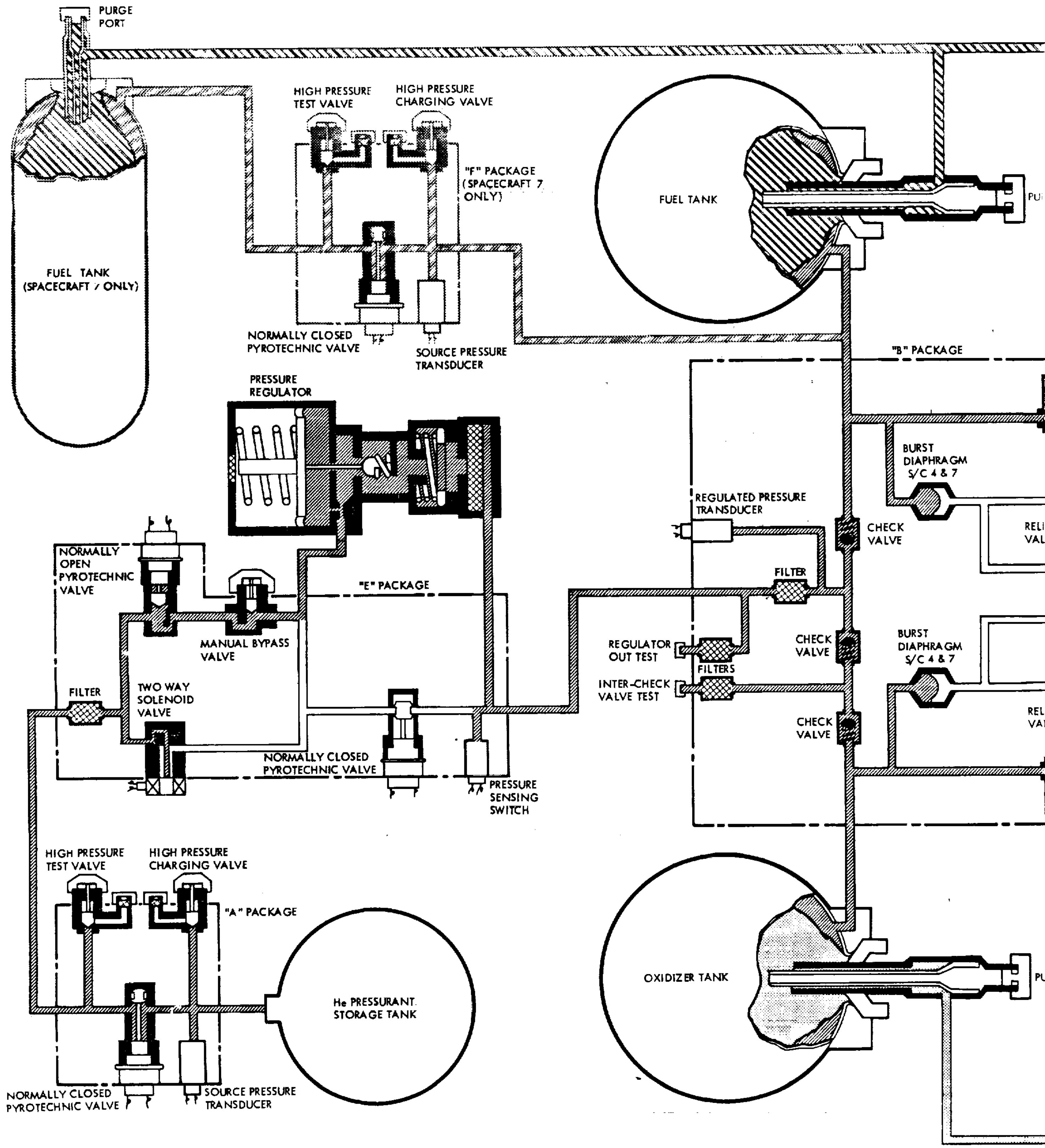


Figure 8-73 Orbit Attitude Maneuvering System Schematic

system to prevent contamination of the system. The pressurant is isolated in the storage tank during pre-launch periods by a normally closed pyrotechnic actuated valve, located in the "A" package. On Spacecraft 7, the pressurant is isolated from the reserve fuel tank by the "F" package.

Fuel/Oxidizer Group

The fuel/oxidizer (propellant) group (Figure 8-73) consists of expulsion bladder storage tanks, "C" and "D" packages and two propellant shut off valves. Charging valves and ports and test valves and ports are provided at accessible points to permit servicing, venting, purging and testing. The propellants are isolated in the storage tanks by normally closed, pyrotechnic actuated valves ("C" and "D" packages). Filters are provided in the "C" and "D" packages, down stream of the isolation valves, to guard against contamination of the thrust chamber assemblies. The propellants used are:

OXIDIZER - nitrogen tetroxide (N_2O_4) conforming to specification MIL - P - 26539 A

FUEL - monomethyl hydrazine ($(CH_3)\ N_2H_3$) conforming to specification MIL - P - 27403

Thrust Chamber Assembly (TCA) Group

The TCA group consists of thrust chambers and electrical solenoid valves. Sixteen TCA's are used per spacecraft (Figure 8-72). Eight twenty-five pound thrust capacity TCA's are used for attitude control, (roll, pitch and yaw). Six one-hundred pound and two eighty-five pound thrust capacity TCA's are used for translational maneuvering.

SYSTEM OPERATION

Pressurant Group

The pressurant tank contains high pressure helium (He) stored at 3000 PSI. (Figure 8-73). The tank is serviced through the "A" package high pressure gas charging port. Pressure from the pressurant tank is isolated from the remainder of the system by a normally closed pyrotechnic actuated isolation valve located in the "A" package. Upon command, the system isolation valve is opened and pressurized helium flows through the "E" package, to the pressure regulator, "B" package and propellant tanks. Normally, pressurant is controlled through system pressure regulator, and regulated pressure flows to the "B" package. The "B" package serves to deliver pressurant at regulated pressure to the fuel and oxidizer tanks, imposing pressure on the propellant tank bladder exteriors. Relief valves in the "B" package prevent over pressurization of the system downstream of the regulator. Burst diaphragms are provided in series with the relief valves, in the "B" package of S/C 4 and 7, to provide a positive leak tight seal between system pressure and the relief valve.

The "E" package provides a secondary mode of pressure regulation in the event of regulator failure. In the event of regulator over-pressure failure, resulting in excess pressure passage through the regulator, a pressure switch ("E" package) intervenes and automatically closes the normally open cartridge valve. Regulated pressure is then controlled manually by the crew by utilizing the OAMS-PULSE switch. Control pressure information is obtained from the "B" package regulated pressure transducer. Should regulator under-pressure failure occur, the crew can manually select the OAMS-REG switch to SQUIB. This selection opens the normally closed valve and closes the normally open valve, thus pressurant by-passes the regulator completely. Pressure is then regulated manually (OAMS-PULSE) by the crew with control pressure information obtained from the "B" package regulated pressure transducer. The "B" package provides a division of pressurant flow to the propellant tanks. The regulated pressure is sensed by the pressure transducer and provides a signal to the cabin instrument, indicating pressure downstream of the regulator. In the event of regulator failure, the crew utilizes the reading to maintain the required pressure in the system for proper operation of pressurant in the propellant tanks. Three check valves prevent back flow of propellant vapors into the pressurant system. The "B" package also affords a safety feature for prevention of over pressure on the fuel and oxidizer tank bladders. Should the system be over pressurized downstream of the regulator, the over pressure would first rupture the burst diaphragms on S/C 4 and 7, then be vented overboard through the relief valves. The relief valves will reset when system pressure returns to normal.

On Spacecraft 7, the pressurant flows from the "B" package to the "F" package. Upon command, the normally closed pyrotechnic valve in the "F" package is opened allowing pressurant to flow to the reserve fuel tank.

Fuel/Oxidizer Group

Fuel and oxidizer are stored in their respective tanks and are isolated from the remainder of the system by normally closed pyrotechnic valves in the "C" (oxidizer) and "D" (fuel) packages. Upon command, the "A" (pressurant), "C" and "D" package isolation valves are opened. The pressurant imposes pressure on the propellant tank bladders and fuel and oxidizer are distributed through their separate tubing manifold systems to the inlet of the thrust chamber solenoid valves. Upon command on Spacecraft 7, the normally closed pyrotechnic valve in the "F" package is opened to allow pressurant to impose pressure on the reserve fuel tank bladder to distribute reserve fuel to the thrust chamber solenoid valve. Two normally open electric-motor valves are located in the propellant feed lines, upstream of the TCA's. In the event of fuel or oxidizer leakage through the TCA solenoid valves, the motor operated valves can be closed by the crew to prevent loss of propellants. The valves can again be actuated open by the crew, when required, to deliver propellants to TCA solenoids.

Thrust Chamber Assembly (TCA) Group

Upon command from the automatic or manual controls, signals are transmitted through the attitude control maneuvering electronics (ACME) to selected TCA's to open simultaneously the normally closed, quick-acting fuel and oxidizer solenoid valves mounted on each TCA. In response to these commands, propellants are directed through small injector jets into the combustion chamber. The controlled fuel and oxidizer impinge on one another, where they ignite hypergolically to burn and create thrust.

SYSTEM UNITS

Pressurant Storage Tank

The helium pressurant is stored in welded, titanium spherical tank. Tank dimension is 16.20 inches outside diameter and has an internal volume of 1696.0 cubic inches. The helium gas is stored at 3000 PSI and held therein by the "A" package normally closed pyrotechnic actuated valve. The pressurized helium is used to expel the fuel and oxidizer from their respective tanks.

Temperature sensors are affixed to the pressurant tank and outlet line to provide readings for the cabin instrument and telemetry.

"A" Package

The "A" package (Figure 8-74) consists of a source pressure transducer, isolation valve, two high pressure gas charging and test valves and filters. The source pressure transducer monitors the pressurant tank pressure and transmits an electric signal to the cabin propellant instrument and spacecraft telemetry system. The normally closed pyrotechnic isolation valve is used to isolate pressure from the remainder of the system. The valve is pyrotechnic actuated to the open position to activate the system for operation. Two dual seal, high pressure gas charging valves and ports are provided, one on each side of the isolation valve. The upstream valve is used for servicing, purging and venting the pressurant tank, while the downstream valve is used to test downstream components. The valve filters prevent contamination during testing and servicing.

"F" Package (Spacecraft 7 only)

The "F" package (Figure 8-74) consists of a source pressure transducer, isolation valve, two high pressure gas charging and test valves and filters. The source pressure transducer monitors the regulated pressure and transmits an electrical signal to the cabin instrument and spacecraft telemetry indicating the amount of regulated pressure for the reserve fuel tank. The normally closed pyrotechnic valve is used to isolate the pressurant from the reserve fuel tank. The valve is pyrotechnic actuated to the open position to activate the reserve fuel system for operation. Two dual seal, high pressure gas charging valves

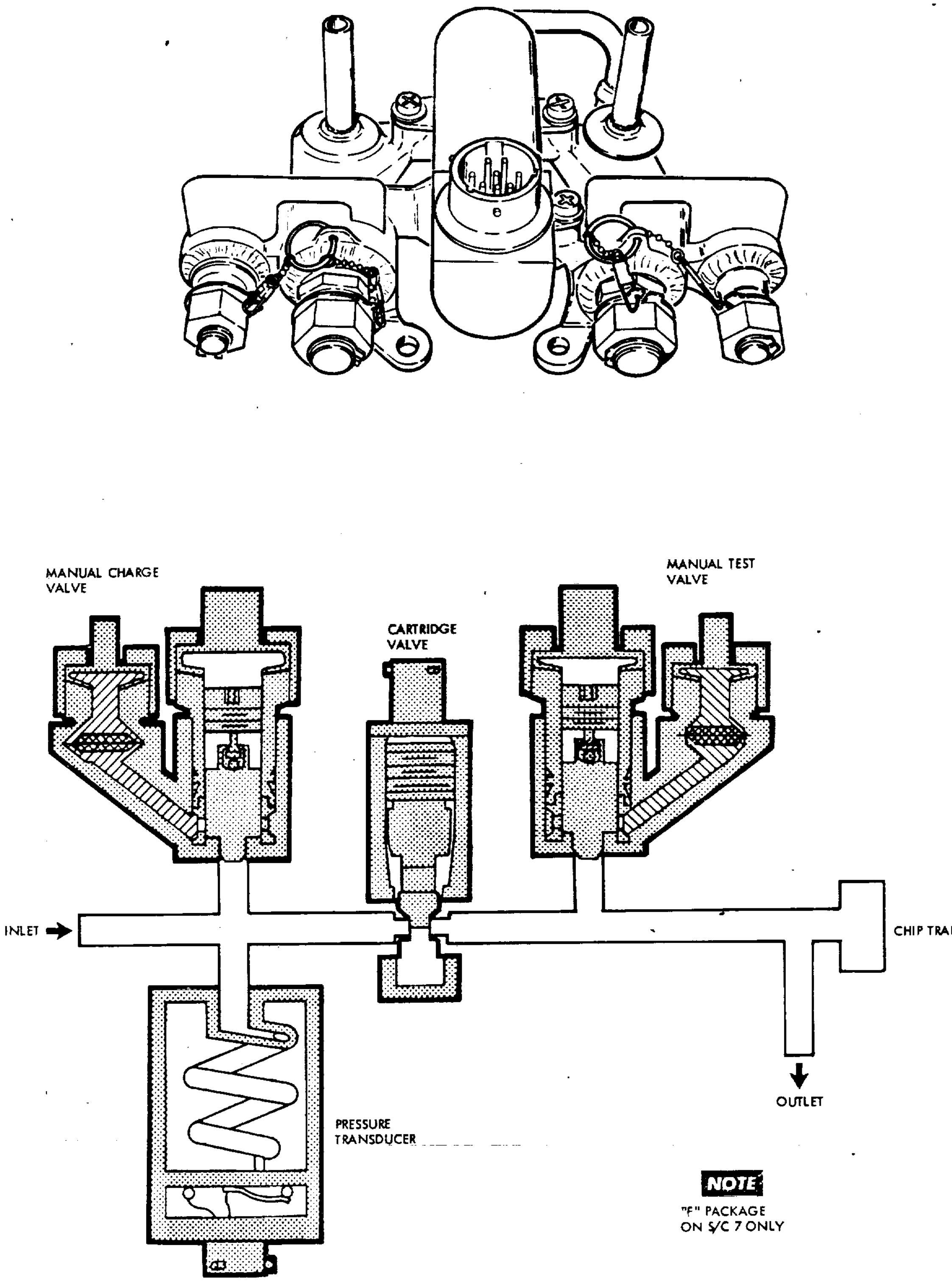


Figure 8-74 OAMS and RCS "A" Package and OAMS "F" Package

and ports are provided, one on each side of the isolation valve. The valve filters prevent contamination during testing and servicing.

"E" Package

The "E" package (Figure 8-75) consists of a filter, one normally open pyrotechnic actuated valve, one normally closed pyrotechnic actuated valve, a normally closed two way solenoid valve, a pressure sensing switch, and a manual by-pass valve. The input filter prevents any contaminants from the "A" package from entering the "E" package. The two pyrotechnic actuated valves are activated (open to closed and closed to open) as required to maintain regulated system pressure, in the event of system regulator malfunction. The two way (open-close) solenoid valve is normally closed and functions upon crew command to maintain regulated system pressure in the event of a system regulator malfunction. The pressure switch senses regulated pressure from the system regulator. Upon sensing over pressure, the pressure switch intervenes and causes the normally open valve to actuate to the closed position, closing the inlet to the pressure regulator. The solenoid valve, when opened, allows pressurant flow through the package after the normally opened valve is actuated to the closed position. The manual by-pass (normally open) test valve is used to divert pressure to the solenoid valve, during system test.

In the normal mode of operation, gas flows through the normally open pyrotechnic valve to the system regulator. In the event system regulator over pressure malfunction, the pressure switch intervenes and causes the normally opened pyrotechnic valve to actuate to the closed position, diverting pressure to the normally closed solenoid valve. The solenoid valve is manually controlled

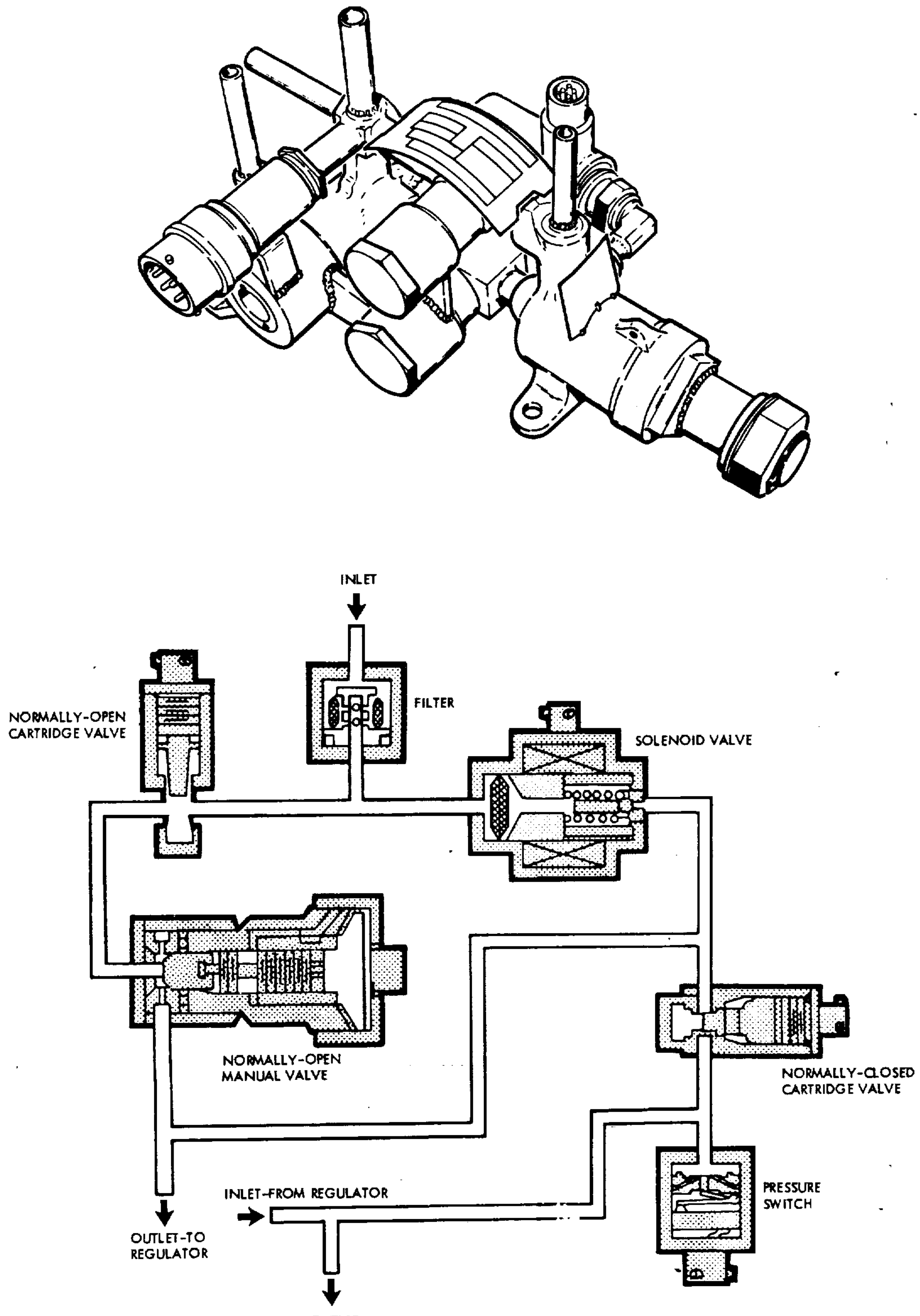


Figure 8-75 OAMS "E" Package

(pulsed) by the crew to maintain regulated system pressure. In the event of system regulator (under pressure) malfunction, the normally closed pyrotechnic valve can be actuated to the open position. Simultaneously insured by the circuitry, the normally open valve is activated to the closed position. This prevents by-pass of the solenoid valve. In this mode,a regulator by-pass circuit is provided and pressure is regulated by the crew.

Pressure Regulator

The pressure regulator (Figure 8-76) is a conventional, mechanical-pneumatic type. The regulator functions to reduce the source pressure to regulated system pressure. An inlet filter is provided to reduce any contaminants in the gas to an acceptable level. An outlet line is provided from the regulated pressure chamber to the pressure switch ("E" package) and activates the switch in the event of an over pressure malfunction.

"B" Package

The "B" package (Figure 8-77) consists of filters, regulated pressure transducer, three check valves, two burst diaphragms, two relief valves, regulator out test port, fuel tank vent valve, inter-check valve test port, oxidizer tank vent valve, and two relief valve test ports. The inlet filter reduces any contaminants in the gas to an acceptable level. Test valve inlet filters prevent any contaminants from entering the system. The regulated pressure transducer monitors the regulated pressure and transmits an electric signal to the cabin instrument and spacecraft telemetry indicating the amount of regulated pressure. A single check valve prevents backflow of fuel vapors into the gas system. Two check valves are provided on the oxidizer side to

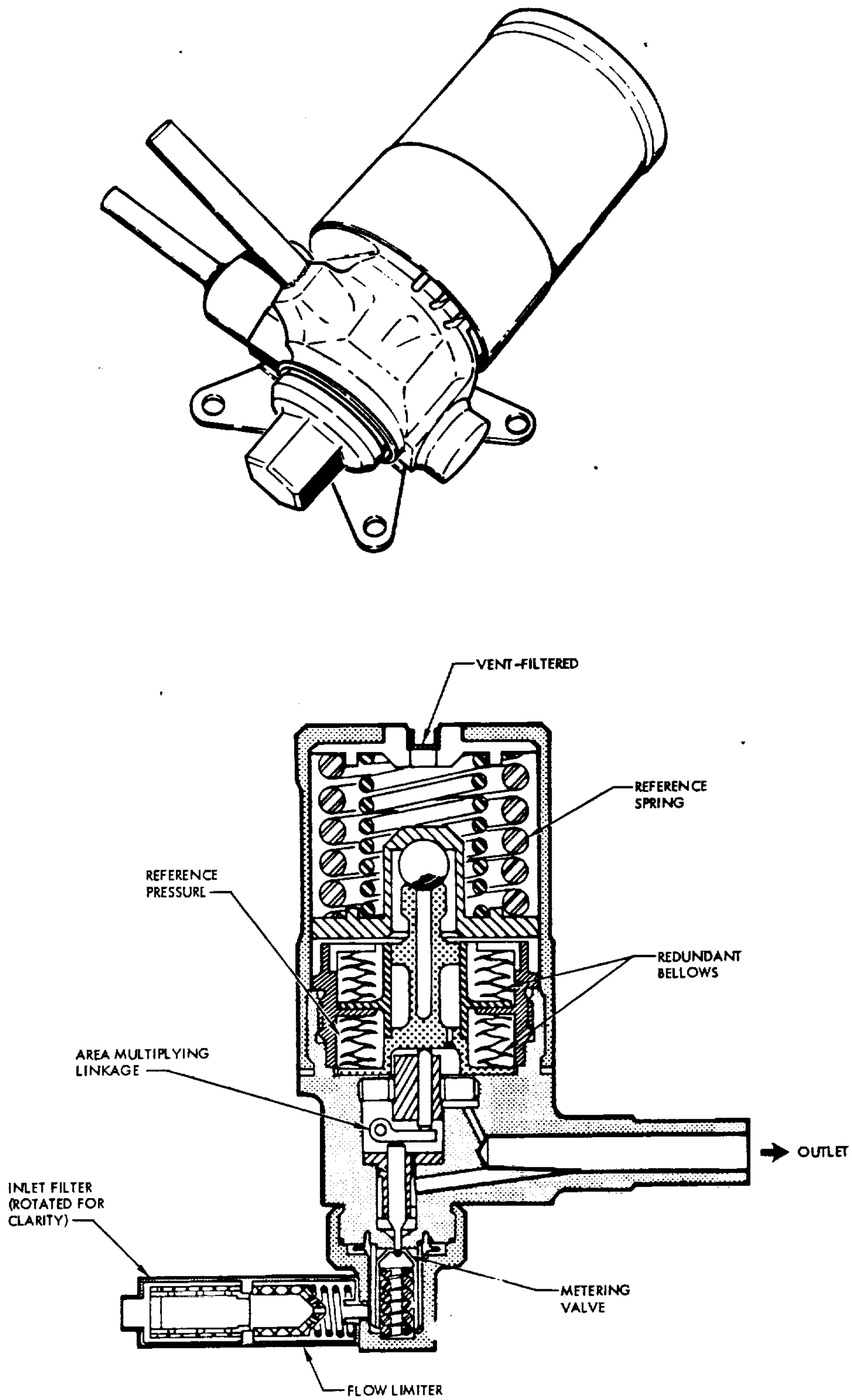
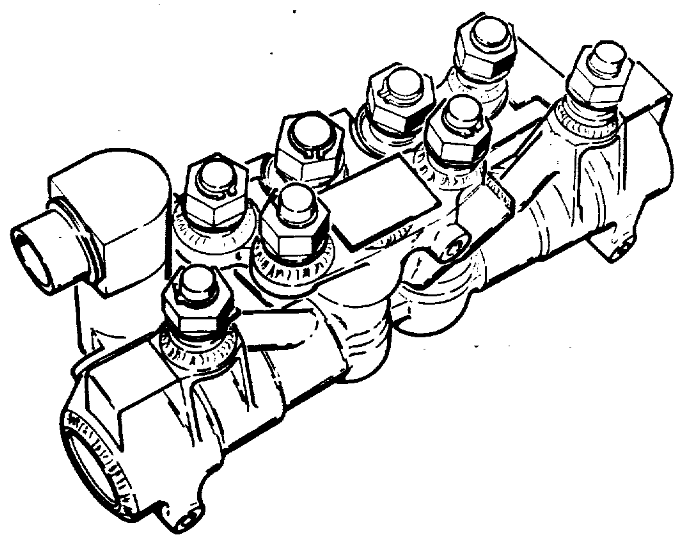


Figure 8-76 OAMS and RCS Pressure Regulator



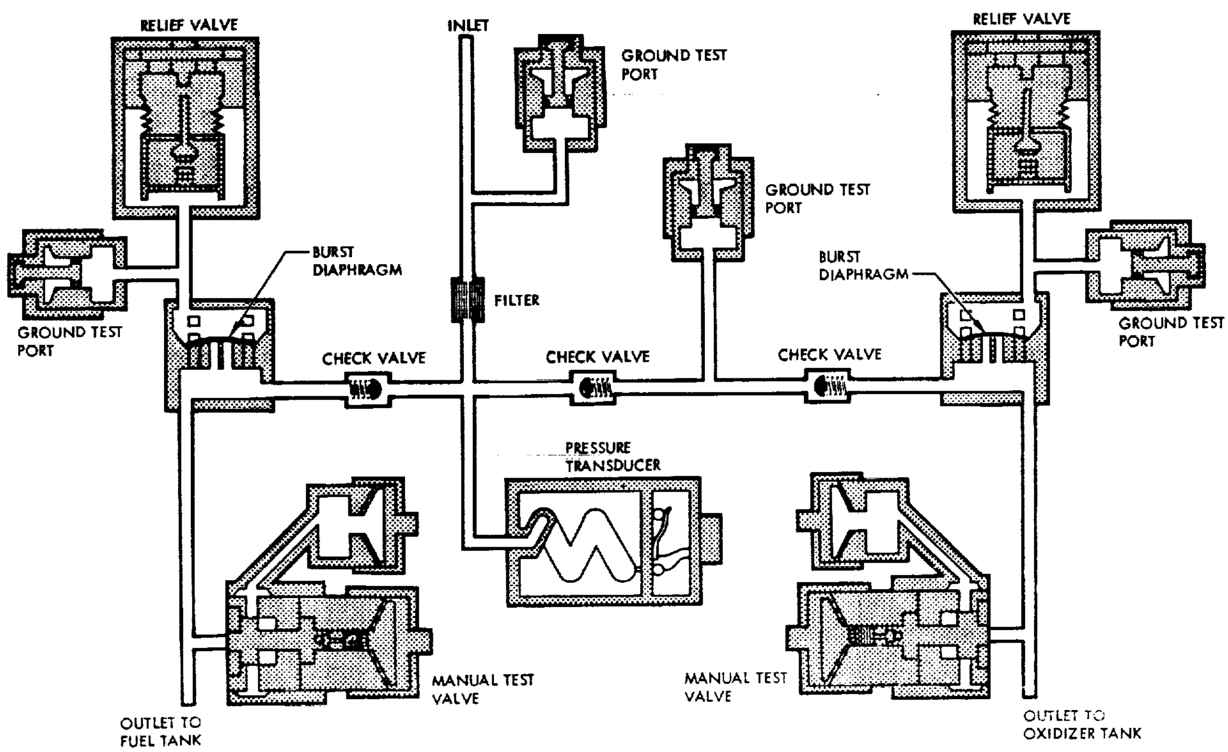


Figure 8-77 OAMS and RCS "B" Package

to prevent backflow of oxidizer into the system. The burst diaphragms are safety (over pressure) devices that rupture when regulated pressure reaches the design failure pressure, thus, prevents imposing excessive pressure on the propellant bladders. The two relief valves are conventional, mechanical-pneumatic type with pre-set opening pressure. In the event of burst diaphragm rupture, the relief valve opens venting excess pressure overboard. The valve reseats to the closed position when a safe pressure level is reached, thereby, prevents venting the entire gas source. Manual valves and ports are provided to vent, purge and test the regulated system.

Fuel Tank

The fuel storage tank (Figure 8-78) is welded, titanium spherical tank which contain an internal bladder and purge port. The tank dimension is 21.13 inches in diameter, and has a fluid volume capacity of 5355.0 cubic inches. The tank bladder is a triple layered Teflon, positive expulsion type. The helium pressurant is imposed on the exterior of the bladder to expel the fuel through the "D" package to the thrust chamber solenoid valves. Purge ports are provided to purge and vent the fuel tank. Temperature sensors are affixed to the input pressurant line, fuel tank exterior and output line to provide readings for the cabin instrument and telemetry.

Reserve Fuel Tank (Spacecraft 7 only)

The fuel tank (Figure 8-86) is a welded, titanium cylindrical tank which contains an internal bladder and purge port. The tank dimension is 5.10 inches outside diameter, 30.7 inches in length and has a fluid volume capacity of 546.0 cubic inches. The helium pressurant is imposed on the exterior of

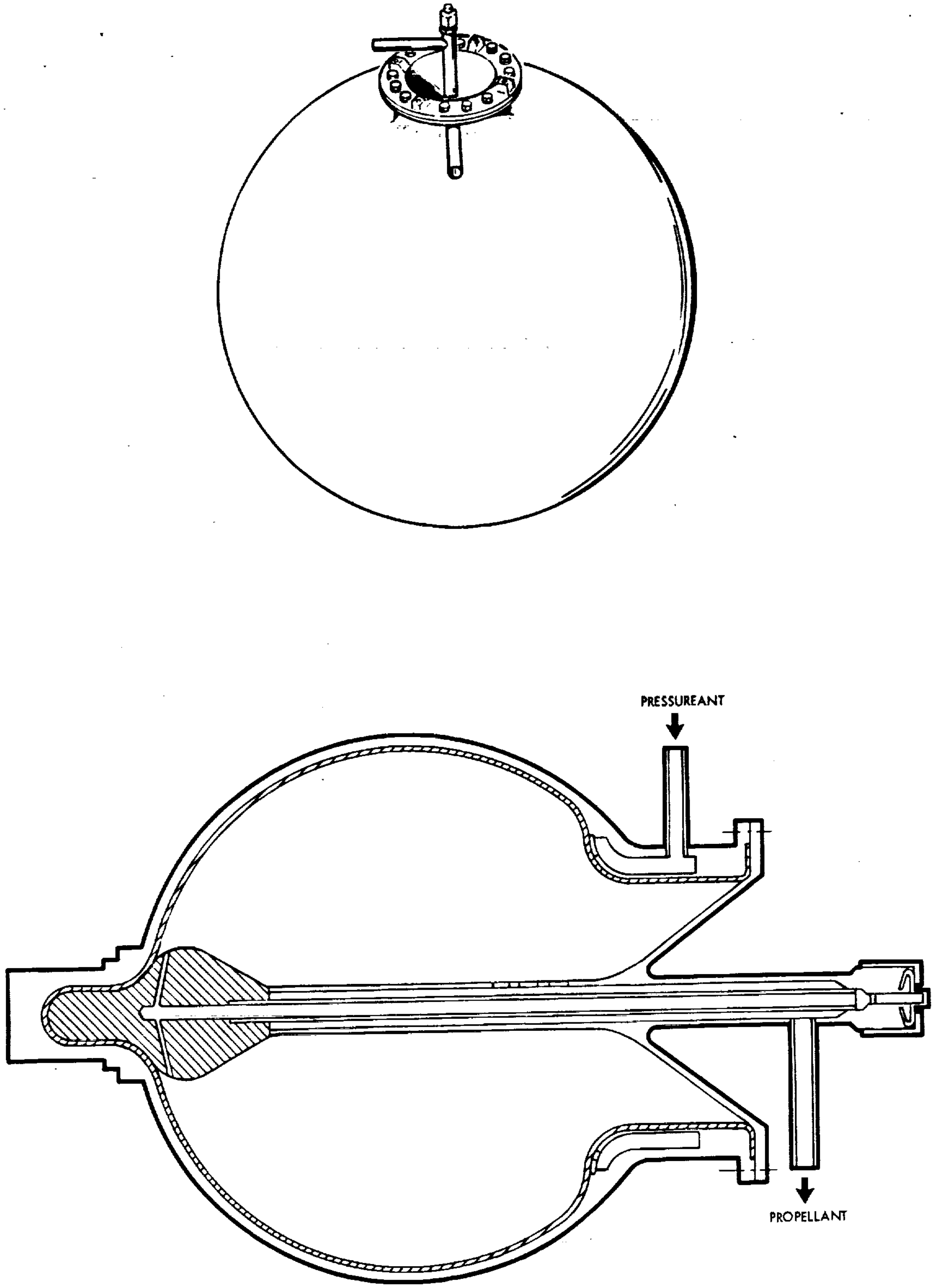


Figure 8-78 OAMS Propellant Tank

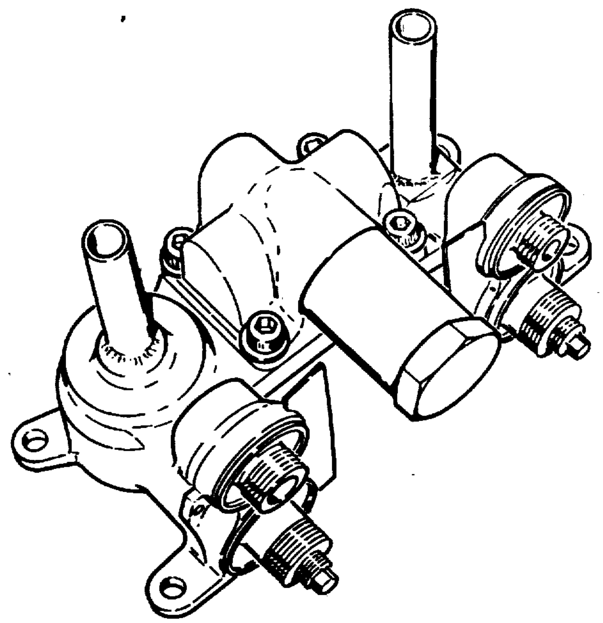
the bladder to expel fuel through the "D" package to the thrust chamber solenoid valves.

Oxidizer Tank

The oxidizer tank (Figure 8-78) is welded, titanium spherical tank which contain a bladder and purge port. The tank dimension is 21.12 inches in diameter, and has a fluid volume capacity of 5355.0 cubic inches. The tank bladder is double layered Teflon, positive expulsion type. The helium pressurant is imposed on the exterior of the bladder to expel the oxidizer through the "C" package to the thrust chamber solenoid valves. Purge ports are provided to purge and vent the oxidizer tanks. Temperature sensors are affixed to the input pressurant line, oxidizer tank exterior and output line to provide readings for the cabin instrument and telemetry.

"C" and "D" Packages

The "C" (oxidizer) and "D" (fuel) packages (Figure 8-79) are identical in function and are located downstream of the tanks of their respective system. Each package consists of a filter, isolation valve, propellant charging valve and test valve. The filter is located at the outlet port to prevent contaminants from entering the downstream system. The normally closed isolation valve is used to isolate propellants from the remainder of the system during the prelaunch waiting period. The isolation valve is pyrotechnic actuated to the open position for system operation. The propellant charging valve is located upstream of the isolation valve and is used for servicing and venting the system. The test valve is located downstream of the isolation valve and is used to test the downstream system.



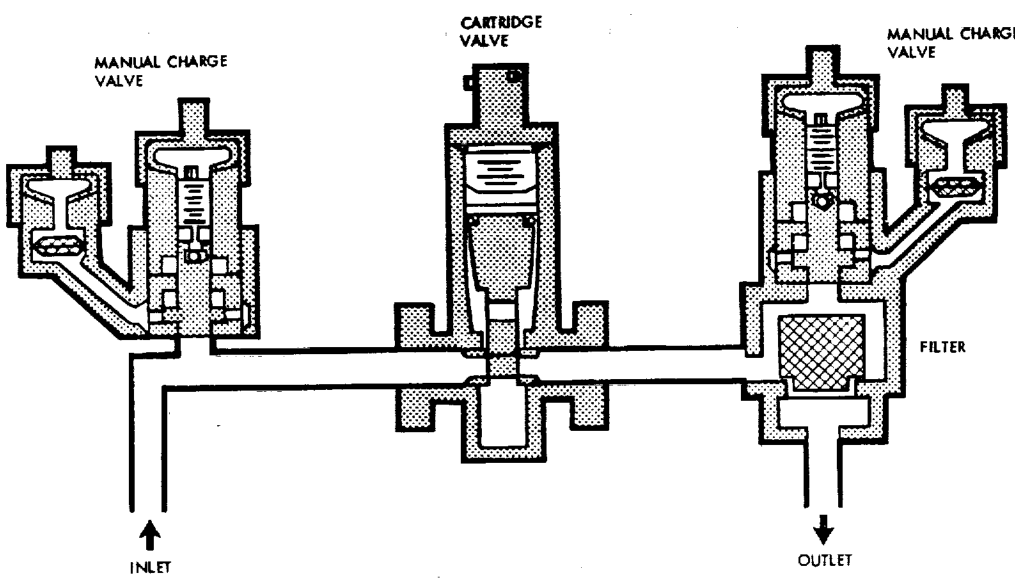


Figure 8-79 OAMS and RCS "C" and "D" Package

Propellant Supply Shutoff/On Valves

Propellant supply shutoff/on valves (Figure 8-80) are provided for both the oxidizer and fuel system and are located downstream of the "C" amd "D" packages in the system. The valves are motor operated, manual/electric controlled type. The propellant valves serve as safeguards in the event of TCA leakage. The valves are normally open, and are closed at the option of the crew to prevent loss of propellants. The valve is thereafter reopened only when it is necessary to actuate the TCA's for the purpose of spacecraft control.

Thrust Chamber Assembly (TCA) Group

Each TCA (Figure 8-81, 8-82 and 8-83) consists of two propellant solenoid valves, an electric heater, injection system, calibrated orifices, combustion chamber and an expansion nozzle. The propellant solenoid valves are quick acting, normally closed, which open simultaneously upon application of an electric signal. This action permits fuel and oxidizer flow to the injector system. The injectors utilize precise jets to impinge fuel and oxidizer streams on one another for controlled mixing and combustion. The calibrated orifices are fixed devices used to control propellant flow. Hypergolic ignition occurs in the combustion chamber. The combustion chamber and expansion nozzle is lined with ablative materials and insulation to absorb and dissipate heat, and control external wall temperature. TCA's are installed within the adapter with the nozzle exits terminating flush with the outer moldline and located at various points about the adapter section suitable for the attitude and maneuvering control required. Electric heaters are installed on the TCA oxidizer valves to prevent the oxidizer from freezing.

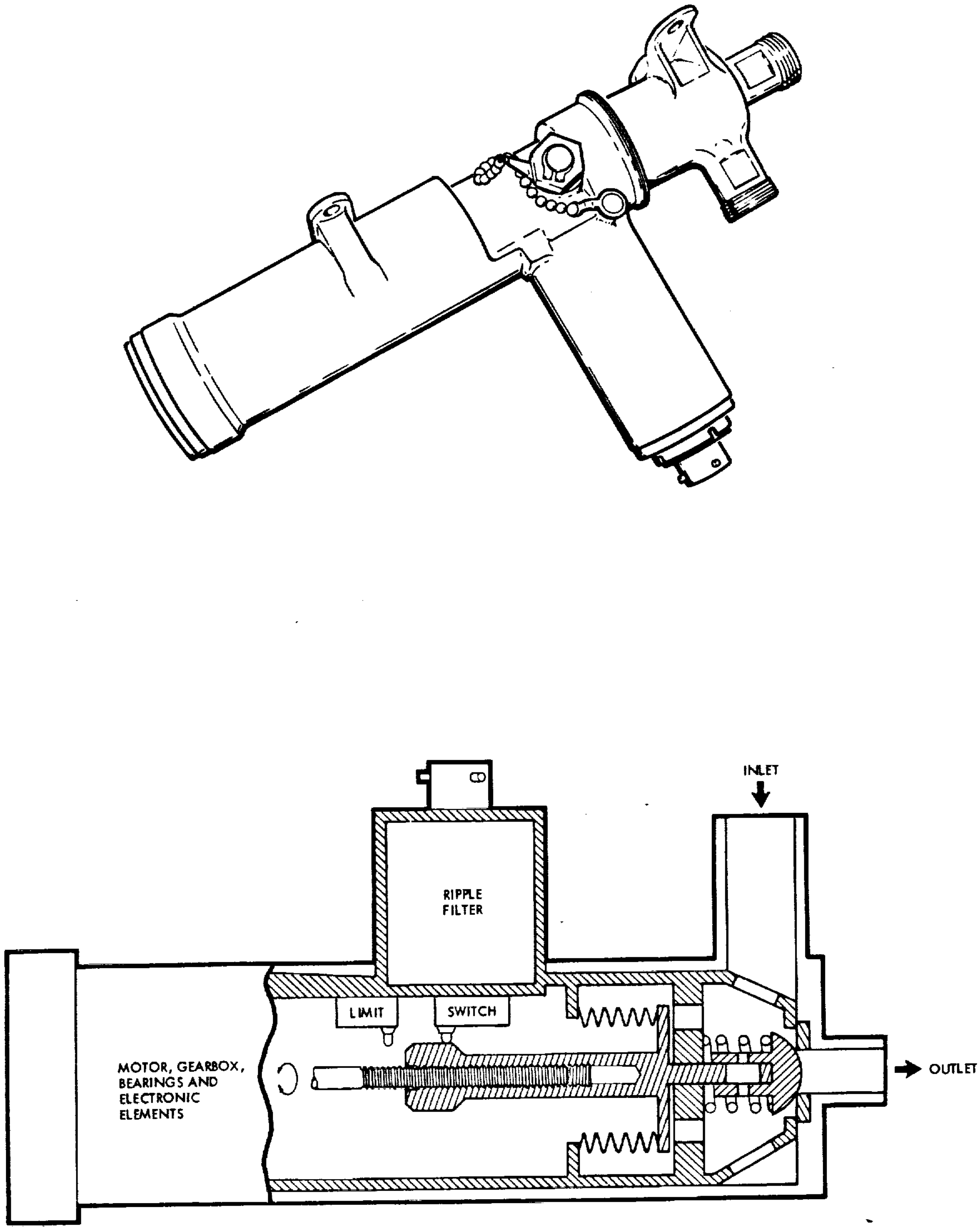


Figure 8-80 OAMS and RCS Propellant Shutoff Valve

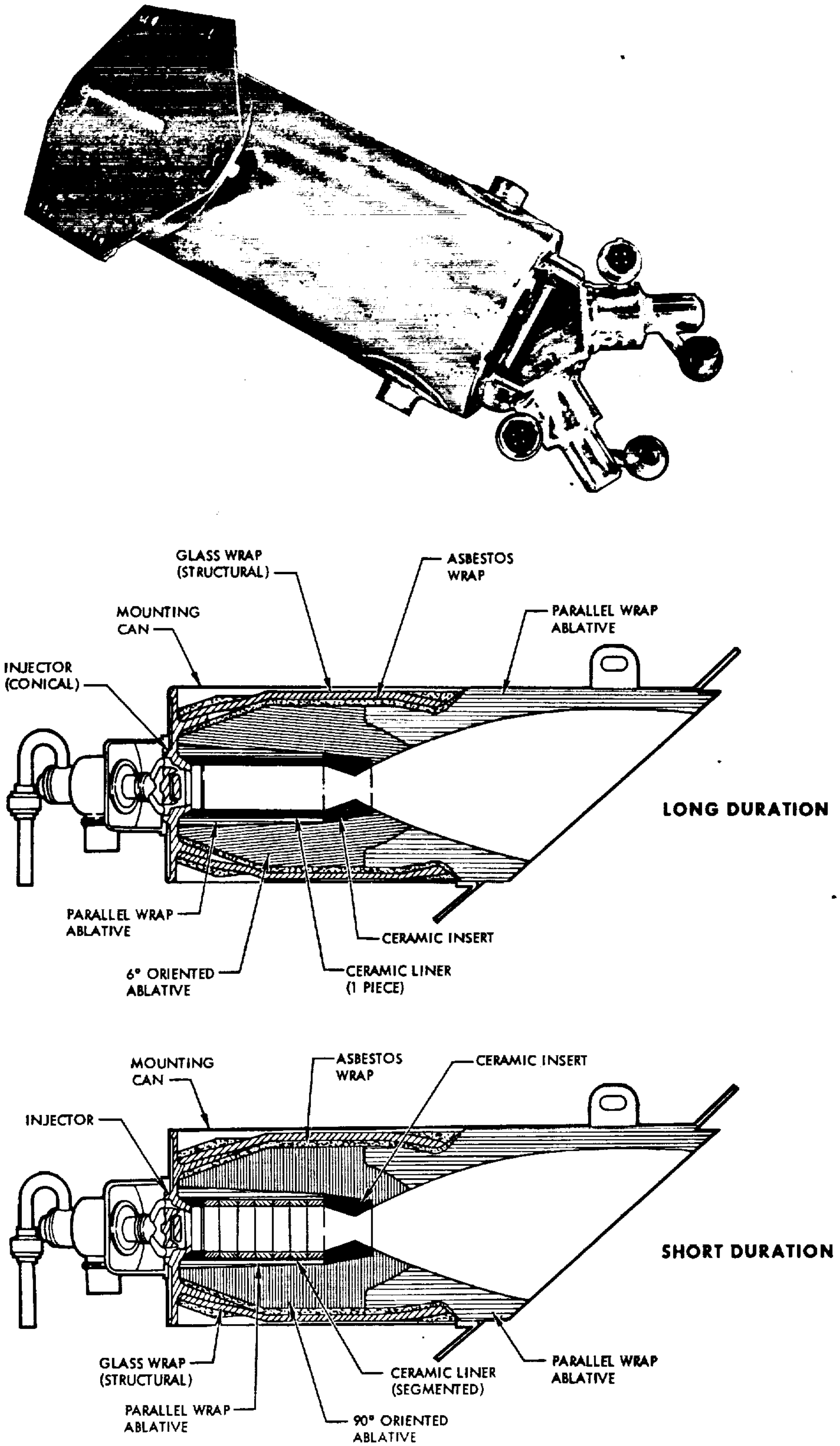
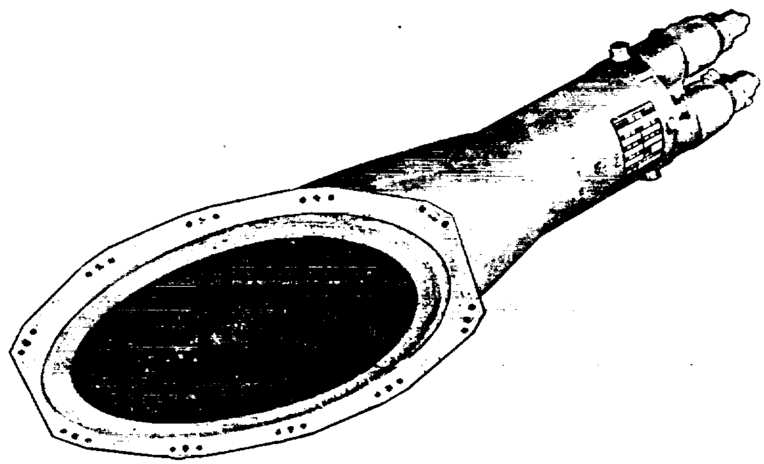
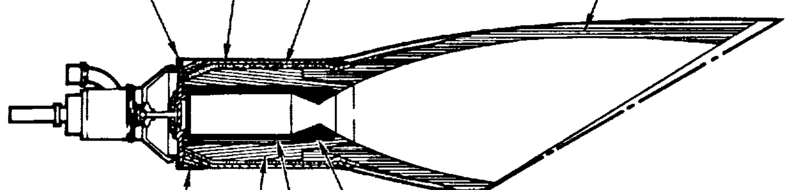
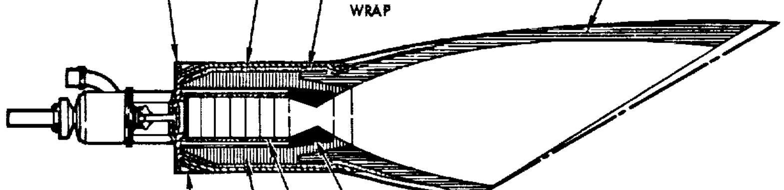


Figure 8-81 OAMS 25 Lb. TCA





LONG DURATION



SHORT DURATION

Figure 8-82 OAMS 85 Lb. TCA

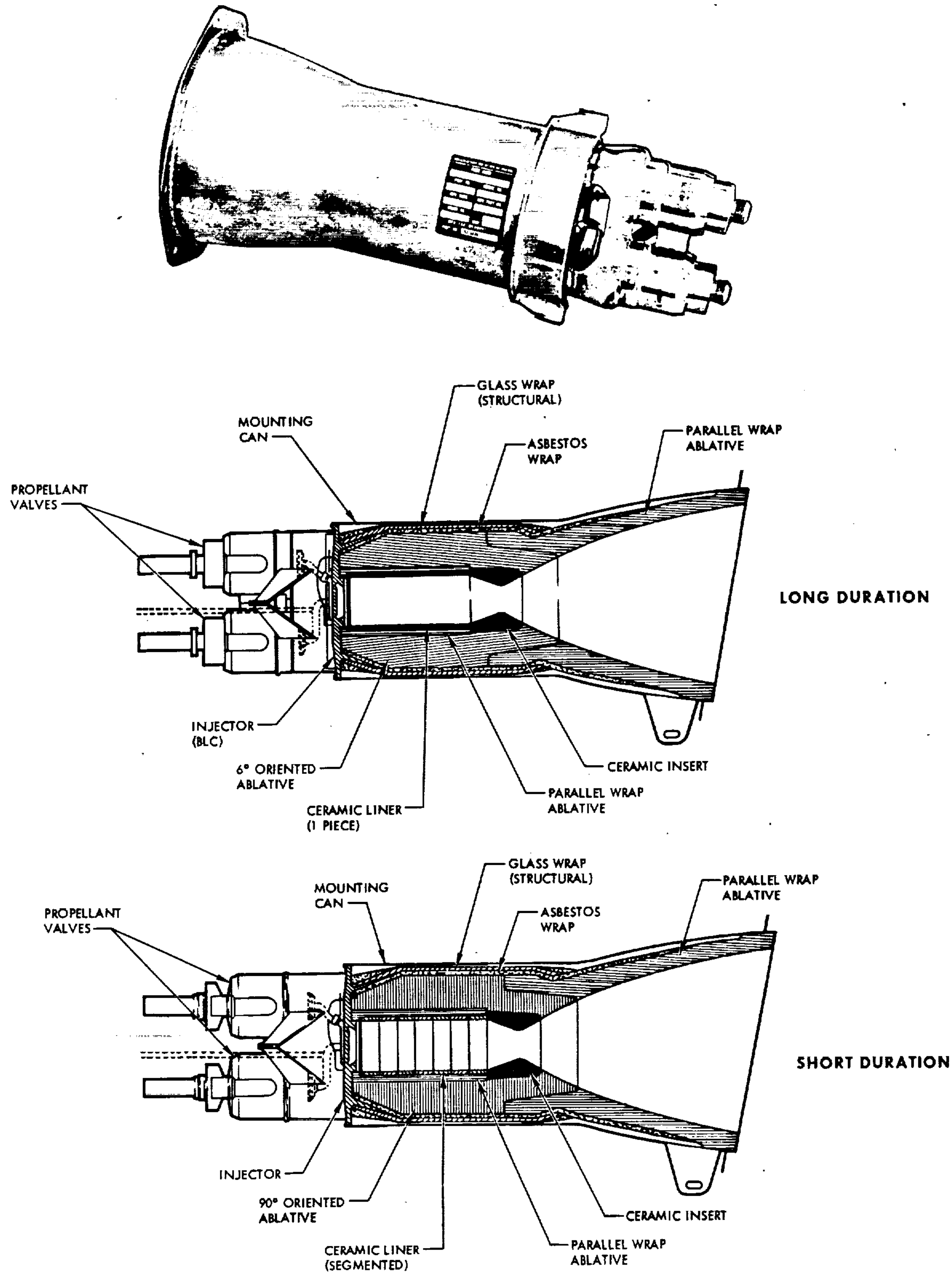


Figure 8-83 OAMS 100 Lb. TCA

Tubing Cutter/Sealer

The tubing cutter/sealer is a pyrotechnic actuated device and serves to positively seal and cut the propellant feed lines. Two such devices are provided for each feed line and are located downstream of the propellant supply on/off valve, one each in the retro and equipment section of the adapter. Prior to retro fire, the equipment section is jettisoned. The devices are actuated to permit separation of the feed lines crossing the parting line, and to contain the propellants upon separation.

RE-ENTRY CONTROL SYSTEM

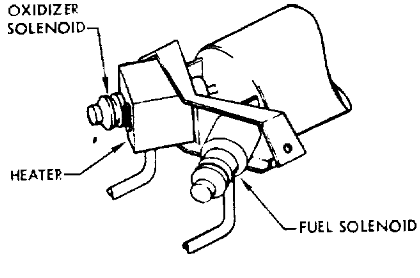
SYSTEM DESCRIPTION

The Re-entry Control System (RCS) (Figure 8-84) is a fixed thrust, cold gas pressurized, storable liquid, hypergolic bi-propellant, self contained propulsion system used to provide attitude control of the spacecraft during re-entry.

NOTE

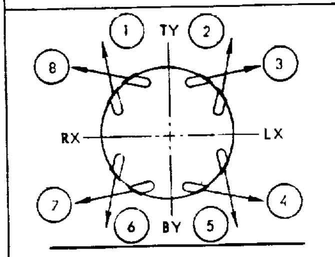
> The RCS consists of two identical but entirely separate and independent systems. The systems may be operated individually or simultaneously. One system will be described, all data is applicable to either system.

The RCS system is capable of operating outside of the earth's atmosphere. Attitude control (roll, pitch and yaw) is obtained by firing the TCA's in groups. The TCA's are mounted at various points about the RCS section of the



DETAIL A

RCS THRUST CHAMBER ATTITUDE CONTROL



THRUST CHAMBER ARRANGEMENT

5	6	PITCH UP
1	2	PITCH DOWN
3	4	YAW RIGHT
7	8	YAW LEFT
3	7	ROLL RIGHT
4	8	ROLL LEFT

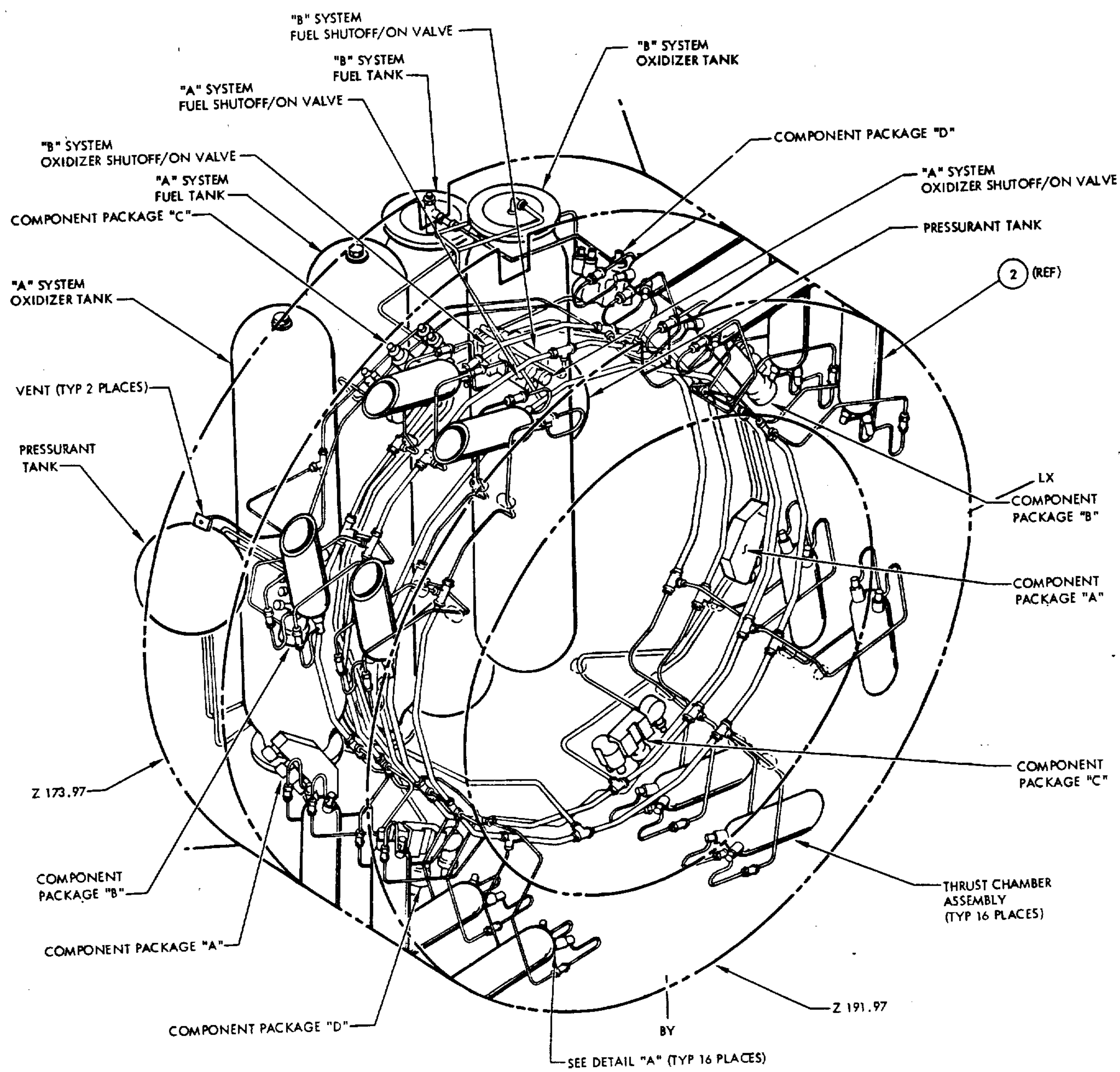


Figure 8-84 Re- entry Control "A" and "B" Systems

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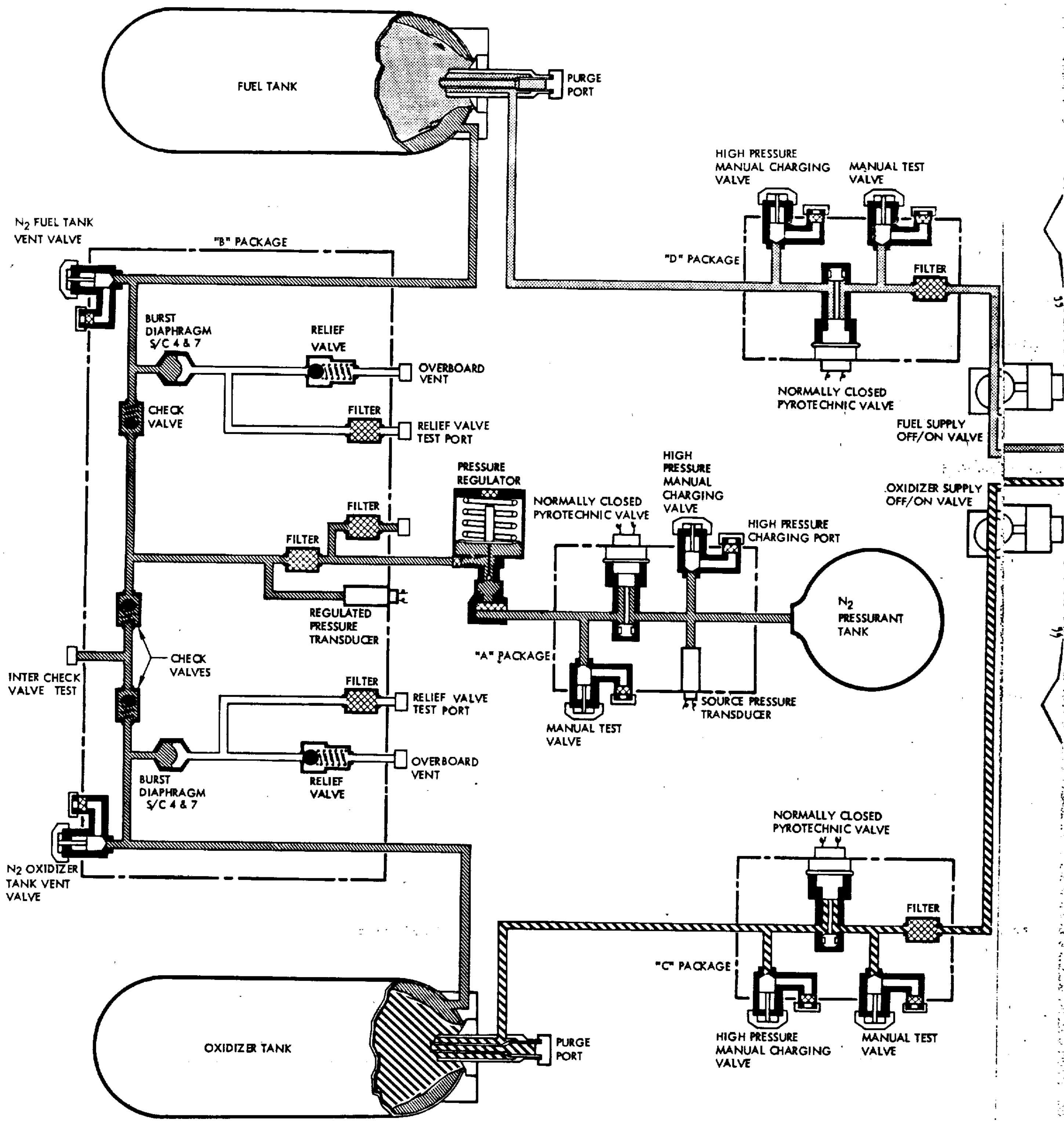

Figure 8-85 RCS (Single System)

Oxidizer - Nitrogen Tetroxide (N_2O_4) conforming to Specification MIL - P - 26539A

FUEL - Monomethyl Hydrazine (CH_3) N_2H_3 conforming to specification MIL - P - 27403

Thrust Chamber Assembly (TCA) Group

The TCA group (Figure 8-84) consists of eight twenty-five pound TCA's used for attitude (roll, pitch and yaw) control of the re-entry module. Each TCA is equipped with thrust chamber and electric controlled solenoid valves. Heaters are provided on the oxidizer solenoid valves to maintain the oxidizer at an operating temperature.

SYSTEM OPERATION

Pressurant Group

(Figure 8-85) High pressure nitrogen (N_2) (pressurant), is stored at 3000 PSI in the pressurant tank. The tank is serviced throught the "A" package high pressure gas charging port. Pressure from the pressurant tank is isolated from the remainder of the system, until ready for operation, by a normally closed pyrotechnic actuated valve located in the "A" package. Stored nitrogen pressure is monitored and transmitted to the cabin instrumentation and spacecraft telemetry system by the source pressure transducer located in the "A" package. Upon command, the "A" package pyrotechnic actuated valve is opened (simultaneously, with propellant "C" and "D" package pyrotechnic actuated valves) and nitrogen flows to the pressure regulator and "B" package. The "B" package provides a division of flow to the propellant tanks. The regulated pressure is sensed by the regulated pressure transducer ("B" package) and provides a signal to the spacecraft telemetry system indicating pressure downstream of the regulator. The check valves prevent backflow of propellant vapors into the pressurant system. The "B" package also provides a safety feature to prevent over pressure of the fuel and oxidizer tank bladders. Should the system be over pressurized downstream of the regulator, the excess pressure is vented over-board through the relief valves. On S/C 4 and 7, the over pressure would first rupture the burst diaphragms, then be vented over-board through the relief valves.

Fuel/Oxidizer Group

Fuel and oxidizer (propellants) are stored in their respective tanks, and are serviced through the high pressure charging ports in the "C" and "D" packages. The propellants are isolated from the remainder of the system, until ready for operation, by the normally closed pyrotechnic valves in the "C" and "D" packages. Upon command, the "A" (pressurant), "C" (oxidizer) and "D" (fuel) package pyrotechnic actuated valves are opened and propellants are distributed through their separate tubing manifold system to the thrust chamber inlet solenoid valves.

Two normally open electric-motor valves are located in the propellant feed lines, upstream of the TCA's. In the event of fuel or oxidizer leakage through the TCA solenoid valves, the motor operated valves can be closed by the crew to prevent loss of propellants. The valves can again be actuated open by the crew, when required, to deliver propellants to the TCA solenoids.

Thrust Chamber Assembly (TCA) Group

Upon command from the automatic or manual controls, signals are transmitted through the Attitude Control Maneuvers Electronics (ACME) to selected TCA's to open simultaneously, the normally closed, quick acting fuel and oxidizer solenoid valves mounted on each TCA. In response to the signals, propellants are directed through small injector jets into the combustion chamber. The controlled fuel and oxidizer impinge on one another, where they ignite hypergolically to burn and create thrust.

SYSTEM UNITS

Pressurant Storage Tank

The nitrogen (N_2) pressurant is stored in a welded, titanium spherical tank. The tank dimension is 7.25 inches outside diameter and has an internal volume of 185.0 cubic inches. Nitrogen gas is stored at 3000 PSI and held therein by the "A" package pyrotechnic valve. This nitrogen under pressure is used to expel the fuel and oxidizer from their respective tanks. Temperature sensors are affixed to the pressurant outlet line to provide readings for the cabin instrument and telemetry.

"A" Package

The "A" package (Figure 8-74) consists of a source pressure transducer, isolation valve, filters and two high pressure gas charging valves. The source pressure transducer monitors the stored pressure and transmits and electric signal to the cabin propellant instrument, indicating the pressure of the stored gas. The normally closed isolation valve is used to isolate the pressure from the remainder of the system.

The valve is pyrotechnically actuated to the open position to activate the system for operation. Two dual seal, high pressure gas charging valves and ports are provided, one on each side of the isolation valve. The upstream valve is used for servicing, venting and purging the pressurant tank, while the downstream valve is used to test downstream components. Filters are provided to prevent contaminants from entering the system.

Pressure Regulator

The pressure regulator is a conventional, mechanical-pneumatic type. The regulator functions to reduce the source pressure to regulated system pressure. An inlet filter is provided to reduce any contaminants in the gas to an acceptable level.

"B" Package

The "B" package (Figure 8-77) consists of filters, regulated pressure transducer, three check valves, two burst diaphragms, two relief valves, regulator output test port, fuel tank vent valve, oxidizer tank vent valve, inter-check valve test port and two relief valve test ports. The inlet filter reduces any contaminants in the gas to an acceptable level. Valve inlet filters prevent contaminants from entering the system. The pressure transducer monitors the regulated pressure and transmits an electrical signal to the spacecraft instrumentation system. A single check valve prevents backflow of fuel vapors into the gas system. Two check valves are provided on the oxidizer side to prevent backflow of oxidizer vapor into the gas system. The burst diaphragms are safety devices that rupture when the regulated pressure reaches the design failure pressure, thus, prevents imposing excessive pressure on the propellant bladders.

The two relief valves are conventional mechanical-pneumatic type with pre-set opening pressure. In the event of burst diaphragm rupture, the relief valve opens venting excess pressure overboard. The valve reseats to the closed position when a safe level is reached, thereby, prevents venting the entire gas source. Manual valves and ports are provided to vent, purge and test the regulated system.

Fuel Tank

The fuel tank (Figure 8-86) is a welded, titanium cylindrical tank which contains an internal bladder and purge port. The tank dimension is 5.10 inches outside diameter, 30.7 inches in length and has a fluid volume capacity of 546.0 cubic inches. The nitrogen pressurant is imposed on the exterior of the bladder to expel fuel through the "D" package to the TCA solenoid valves. The purge port is provided to purge and vent the fuel tank bladder. Temperature sensors are affixed to the nitrogen input line and fuel output line to transmit signals to telemetry stations.

Oxidizer Tank

The oxidizer tank (Figure 8-86) is a welded, titanium cylindrical tank which contains a bladder and purge port. The tank dimension is 5.10 inches outside diameter, 25.2 inches in length and has a fluid volume capacity of 439.0 cubic inches. The bladder is a double layered Teflon, positive expulsion type. The nitrogen pressurant is imposed on the exterior of the bladder to expel the oxidizer through the "C" package to the TCA solenoid valve. The purge port is provided for purging and venting the oxidizer tank bladder. Temperature sensors are affixed to the nitrogen input line and oxidizer output line to trans-

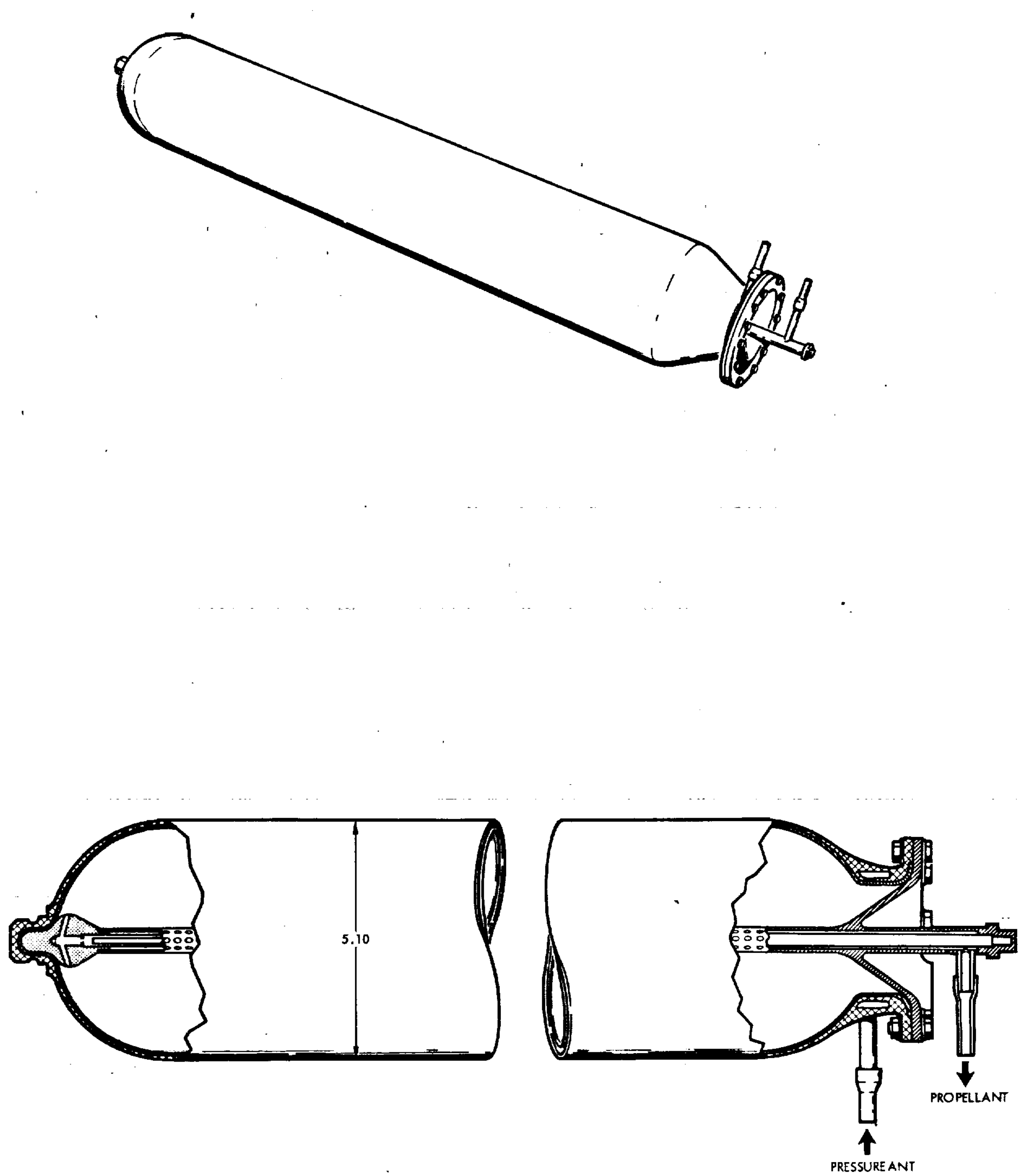

Figure 8-86 RCS Propellant Tanks

mit signals to telemetry stations.

"C" and "D" Packages

The "C" and "D" packages (Figure 8-79) are identical in function and are located downstream of the tanks of their respective system. Each package consists of filters, an isolation valve, propellant charging valve and test valve. The filter located at outlet port reduces contaminants to an acceptable level. The valve and port filters prevent contaminants from entering the downstream system. The normally closed isolation valve is used to isolate propellants from the remainder of the system during the pre-launch waiting period. The isolation valve is pyrotechnic actuated to the open position for system operation. The propellant charging valve is located upstream of the isolation valve and is used for servicing and venting the system. The test valve is located downstream of the isolation valve and is used to test the downstream system.

Propellant Supply Shutoff/On Valves

Propellant supply shutoff/on valves (Figure 8-80) are provided for both the oxidizer and fuel system, and are located downstream of the "C" and "D" packages in the system. The valves are motor operated, manual/electric controlled type. The valves are normally open, and are closed at the option of the crew to prevent loss of propellants. The valves are reopened only when the TCA's are needed for spacecraft control.

Thrust Chamber Assembly (TCA) Group

Each TCA (Figure 8-87) consists of two propellant valves, injection system, calibrated orifices, combustion chamber and expansion nozzle. The fuel and

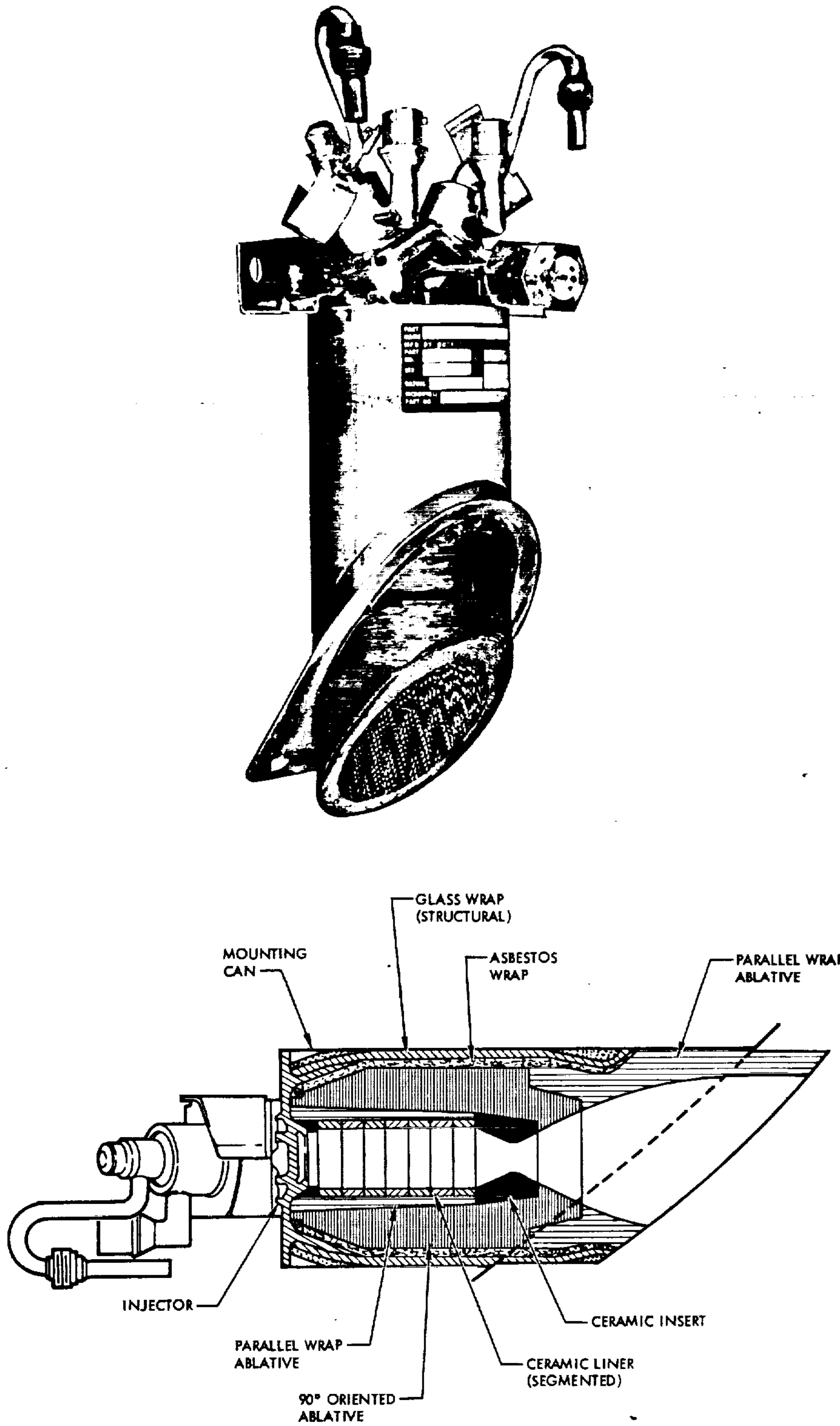

Figure 8-87 RCS 25 Lb. TCA

oxidizer solenoid valves are quick acting, normally closed, which open simultaneously upon application of an electric signal. The action permits fuel and oxidizer flow into the injector system. The injectors use precise jets to impinge fuel and oxidizer streams on one another for controlled mixing and combustion. The calibrated orifices are fixed devices used to control propellant flow. Hypergolic ignition occurs in the combustion chamber. The combustion chamber and expansion nozzle is lined with ablative materials and insulation to absorb and dissipate heat and control external wall temperature. TCA's are installed within the RCS section mold line, with the nozzles terminating flush with the outer mold line. TCA's are located at fixed points in the RCS section in a location suitable for attitude control. Electric heaters, located on the oxidizer valve, are used to prevent the oxidizer from freezing.